FTD-HC-23-374-71

AD 740211

FOREIGN TECHNOLOGY DIVISION

AIR FORCE SYSTEMS COMMAND

AIRCRAFT POWERPLANTS. SYSTEMS AND COMPONENTS

by

N. T. Domotenko, A. S. Kravets,
A. I. Pugachev, T. I. Sivashenko

FOREIGN TECHNOLOGY DIVISION

Approved for public release;
Distribution unlimited.

460

UNCLASSIFIED

Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Handbook						
Aircraft Engine						
Aircraft Hydraulic Equipment						
Aircraft Deicing Equipment						
Fire Control Equipment						
Engine Exhaust System						
Aircraft Propeller						
Engine Starter System						
Engine Control System						
Automatic Control System						

UNCLASSIFIED

Security Classification

UNCLASSIFIED

Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) Foreign Technology Division Air Force Systems Command U. S. Air Force	2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED 2b. GROUP

3. REPORT TITLE

AIRCRAFT POWERPLANTS. SYSTEMS AND COMPONENTS

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

Translation

5. AUTHOR(S) (First name, middle initial, last name)

Domotenko, N. T.; Kravets, A. S.; Pugachev, A. I.; Sivashenko, T. I.

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
1970	458	62

8a. CONTRACT OR GRANT NO. b. PROJECT NO. 3066 c. d.	9a. ORIGINATOR'S REPORT NUMBER(S) FTD-HC-23-374-71 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)

10. DISTRIBUTION STATEMENT

Approved for public release; distribution unlimited.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY Foreign Technology Division Wright-Patterson AFB, Ohio

13. ABSTRACT

The book deals with general problems of layout and operation of equipment of aircraft power plants. Given are methods for calculation of systems and devices. The assumptions accepted for educational purposes, which hardly effect the accuracy of results, enabled the authors to obtain comparatively simple formulas and graphs for checking the operation of systems and devices and their design. The book represents a text book for students of civil aviation educational institutions. It can be used also by engineering technical personnel of operational and maintenance enterprises of civil aviation, factories of the aircraft industry, as well as students of aviation institutes. ( )[AM0143199]

DD FORM 1 NOV 65 1473

UNCLASSIFIED

Security Classification

FTD -HC-23-374-71

EDITED TRANSLATION

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By: N. Domotenko, A. S. Kravets, A. I. Pugachev, T. I. Sivashenko

English pages: 458

Source: Aviatsionnyye Silovyye Ustanovki. Sistemy i Ustroystva 1970, pp. 1-352

Translated under: F33657-71-D-0057

UR/0000-70-000-000

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INTRODUCTION

The aircraft powerplant is designed to create a thrust force, and consists of the engine and also the systems and components which support operation of the engine.

The powerplant includes the following systems: fuel, oil, cooling, fire protection, anti-icing, starting, induction, and exhaust; the components include: engine mounting, nacelle, powerplant controls, and propeller (for piston and turboprop engines). This division is somewhat arbitrary, since the listed systems and components are intimately related with one another on the flight vehicle.

The powerplant systems and components are usually considered as powerplant equipment. Since engine theory and design are covered in specialized courses, in the present book the aircraft powerplants are studied from the viewpoint of the principles involved in integrating their equipment, the actual hardware involved, and analysis of the operation of the systems and components which make up this equipment.

The resolution of the problems associated with increasing flight speed, range, altitude, and safety depends not only on the thrust (power), ceiling, and economy of the engine, but also on how successfully the problems of integrating the aircraft powerplant equipment are solved. The general requirements imposed on aircraft powerplant equipment are:

ensuring flight, maintenance, and economical performance of the flight vehicle;

reliability, safety, survivability;

manufacturing simplicity and operational convenience;

long service life;

low weight and small size;

low aerodynamic and hydraulic resistances.

In studying each type of equipment in the present book, we examine to the extent possible the specific requirements and means for their realization.

The development of powerplant equipment is inseparably linked with the development of aircraft engineering. Soviet technical aviation ideas have contributed to the considerable growth of aviation science in general and, in particular, to the development of aircraft powerplants and their equipment. Proof of this includes the powerplants of record-setting airplanes and helicopters, the extensive use of Soviet flight vehicles and engines, not only in the USSR but also abroad, and the successes in the conquest of space.

The definition of the "system" concept is very important for a clear-cut classification of the systems used in aircraft powerplants. In the aircraft powerplant course we take a <u>system</u> to mean an ensemble of subsystems which are combined, have a common function, and support the operation of the engine. The characteristic features of such a system are the presence therein of a working body and its motion through the subsystems during the time the engine is operating. Depending on the properties of the working body, the systems may be hydraulic, gaseous, electrical. In turn, with regard to type of liquid the hydraulic systems are subdivided into fuel, oil, water systems, and the gaseous systems are divided into air, nitrogen, and so on.

A <u>subsystem</u> consists of assemblies, lines, and fittings (connections). By assembly, we mean a machine, instrument, mechanism, and so on. With regard to purpose, assemblies are divided into specialized items in the form of tanks, bottles, pumps, filters, radiators, coolers, heaters, regulating and controlling equipment.

Depending on the working body flow scheme, the systems used in aircraft powerplants can be open or closed (circulating). In the open systems the working medium travels from the source 1 (Figure 1a) to the user 3 along the pressure (suction, delivery) line 2. In the closed systems the working medium travels from the source to the user along the pressure line and then returns to the source through the return lines 4 and 5 or is directed to the pressure line. Systems are encountered in which part of the working medium is returned to the source and part is directed to the pressure line. Depending on these schemes, the closed systems are divided into:

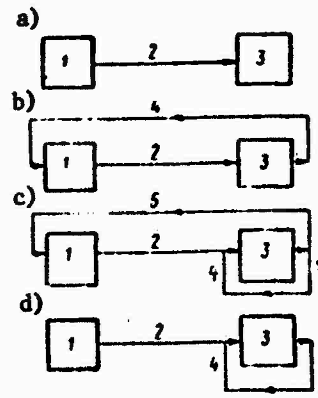


Figure 1. Classification of powerplant systems

single-contour, if the working medium returns to the source (Figure 1b);

two-contour, if part of the working medium is returned to the source and part is directed into the pressure line (Figure 1c);

short-circuit, if the working medium does not return to the source but is redirected into the pressure line leading to the user (Figure 1d).

In view of the fact that many systems are hydraulic, the general questions of system hydraulics are discussed in a separate chapter.

CHAPTER I

AIRCRAFT POWERPLANT CONFIGURATION AND MOUNTING

Classification of Aircraft Powerplants

Aircraft powerplants are classified on the basis of various characteristics, including the types of engines used. Civil airplanes and helicopters in the USSR use powerplants with piston engines (PE), turboprop engines (TPE), and turbojet engines (TJE). Very promising are the powerplants with ramjet engines (RJE) and rocket engines (RE) (Figures 2 and 3), including the liquid rocket engines (LRE).

Up until the mid-1940's the piston engine was the basic aircraft engine type and some airplanes reached speeds of 190 - 310 m/sec and altitudes of 12,000 - 13,000 m using these engines. The thrust P developed by a piston powerplant is expressed as

$$P = \frac{N\eta_p}{V}\ N,$$

where N — is the effective engine power in W;

η_p — is the propeller efficiency;

V — is the flight velocity in m/sec.

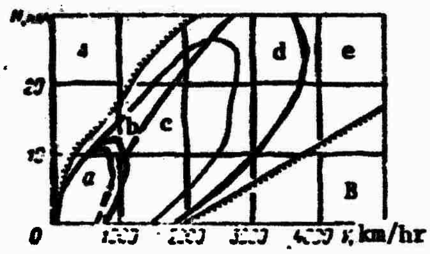


Figure 2. Regions of rational application of different engine types:

a) PE; b) TPE; c) TJE; d) RJE; e) RE; A) region of insufficient wing lift for airplane; B) region of high drag

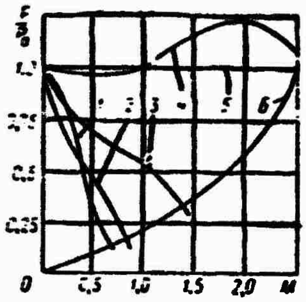


Figure 3. Variation of relative thrust of different engine types with flight Mach number:

1) PE; 2) TPE; 3) TPE; 4) TJE; 5) LRE; 6) RJE

Since the effective power of the PE is independent of the flight speed and the prop efficiency decreases sharply with approach to a Mach number of one, we see from the formula that the thrust available decreases rapidly with increase of the flight speed. At the same time the thrust required increases markedly with increase of the flight speed, and at high speeds a large increase of PE power is required, which leads to considerable increase of PE weight and size.

Supersonic speeds and high flight altitudes were reached when the jet engines appeared. In contrast with the engines using propeller thrust (indirect reaction engines), the jet engines (direct reaction engines) make it possible to concentrate in a single unit large thrust power along with considerably smaller size and less weight in comparison with the PE. At flight Mach numbers $M < 1$ the thrust of the TJE is nearly independent of the flight speed, and for $M > 1$ it increases with increase of the speed.

The TPE is widely used at subsonic speeds (165 - 220 m/sec) and moderate flight altitudes (6,000 - 10,000 m) on passenger and transport airplanes. The primary advantages of these engines are: the possibility of obtaining high power in a single unit (more than 11,000 kW); low specific weight and high power per unit frontal area, low fuel consumption (for flight speeds higher than 210 m/sec the fuel consumption of the TPE is less than that of the PE); the possibility of using the propellers for deceleration to shorten markedly the airplane landing ground roll distance.

The high takeoff thrust and possibility of using propeller reversing during the landing roll improve the takeoff and landing characteristics of airplanes with TPE. This makes it possible to operate them from smaller fields than airplanes with TJE. The economy of airplanes with TPE is better than that of airplanes with TJE at flight speeds up to 365 m/sec, and for the same load capacity the former provide considerably greater flight range. The primary drawbacks of these engines in comparison with the TJE are higher specific weight, higher maintenance costs, lower reliability because of the presence of the propeller, and the limited installation possibilities.

Use of the TJE is advantageous at flight speeds of 200 m/sec and higher. The TJE combines in itself the engine and propulsor, has low specific weight, small size, is simple in construction, and can develop high thrust (180 KN or more). The absence of a propeller facilitates considerably the TJE powerplant installation and reduces drag. This is the primary type of engine for flight at supersonic speeds.

At intermediate and transonic flight speeds the most economical engine is the bypass turbojet or, as it is sometimes called, the turbofan engine (TFE). Under certain conditions the TFE has some advantages over the TJE at supersonic flight speeds. These engines are also used to drive helicopter main rotors, and they are the preferred type of engine for vertical takeoff and landing (VTOL) airplanes. In spite of the differences in configuration, all engines of this type have one common feature: the presence of a second (outer) flow path in which the flow bypasses the combustion chamber and turbine and then mixes with the combustion products of the primary flow path, increasing their mass and reducing the propulsive jet velocity.

In comparison with the TJE these engines have lower fuel consumption (by 5 - 12%) and lower weight for the same thrust, better takeoff characteristics (they develop considerably greater thrust), and lower noise level (by 10 - 15 dB) (in comparison with the noise level of the conventional TJE without silencer).

The thrust of the FJE consists of the thrust created by the primary (inner) flow and the thrust created by the bypass (outer) flow. An important characteristic of the FJE is the bypass ratio, i.e., the ratio of the air mass flowrate through the secondary flow path: $m = \frac{G_{a_2}}{G_{a_1}}$, whose value for the modern TFE varies from 0.23 to 3.5 or more.

When a TFE is installed on an airplane there is an increase of the diametral dimensions of the air intakes or inlet area, particularly for large bypass ratios.

The engine thrust efficiency is defined by the formula

$$\eta_p = \frac{2}{1+\frac{V_e}{V}},$$

where V_j — is the propulsive jet velocity;

V — is the flight speed.

It is obvious that the propulsive jet velocity is an important parameter in obtaining high values of ηp. As the bypass ratio increases the discharge velocity decreases; the thrust efficiency increases and reaches a maximal value at a flight speed equal to the propulsive jet exhaust velocity.

Replacing a TJE by a TFE makes is possible to increase the payload for a given flight range (as a result of reduction of the required fuel supply), reduce the specific fuel consumption and the noise level, and also shorten the takeoff distance for a given airplane weight.

At supersonic flight speeds $M \geq 2.5$ the RJE has the best economy. The economy and specific thrust of the RJE at $M = 3 - 3.5$ reach and even exceed the comparable characteristics of the afterburning turbojet engine (ATJE). These engines are also used with success at lower flight speeds because of their low specific weight and simplicity of construction. But the impossibility of independent takeoff and low efficiency at low flight speeds are significant drawbacks of the RJE. Therefore the RJE is encountered only in combination with other types of engines, or the acceleration of the RJE powered airplane is accomplished by means of the powerplant thrust of a carrier airplane. The use of the RJE is possible only in combination with other types of engines to provide for airplane takeoff and acceleration to the design flight conditions.

Rocket engines surpass all other types of aeroengines in magnitude of thrust developed. An important characteristic of these engines is that their thrust is independent of altitude; they also have low specific weight and high efficiency at supersonic flight speeds. All this permits using solid and liquid rocket engines for attaining high flight altitudes and speeds. These engines are used with success for launching rockets for various purposes, including intercontinental ballistic and space rockets. The high fuel consumptions limit the flight time for continuous operation of the solid and liquid rocket engines. On some flight vehicles they are used as boosters to increase the thrust briefly during takeoff or in flight. These engines are not yet used on passenger airplanes. However, successful flights of experimental airplanes with such engines indicate their rapid improvement and the possibility of their use on transport airplanes.

Number of Engines

The specifications for civil airplanes specify: the cruising speed V_{cr} at the design altitude, the takeoff ground roll L_{gr} under standard atmospheric conditions ($p = 760$ mm Hg, $t = 15°$ C, $\rho = 0.125$ kgf $\cdot$ sec^2/m^4), and also the minimal permissible rate of climb V_y or the minimal climb angle θ with one engine out.

On this basis, the available thrust of the engines installed on the airplane is determined by the following conditions:

provide airplane takeoff from a runway of given length;

possibility of continued flight and climb with one engine out;

a given noise level in the airport area during takeoff (Chapter IX).

The thrust (power) P_{to} (N_{to}) required for an airplane to take off from a runway of given length can be found from the value of P_{av}:

for airplanes with TJE

$$P_{взл} = \frac{P_{ср}}{0{,}93};$$

for airplanes with PE and TPE

$$N_{взл} = 107\,P_{ср}.$$

The average value of the thrust P_{av} during the takeoff ground roll from the moment the airplane begins to move until it reaches the unstick speed is found from the equation

$$P_{ср} = G_{взл}\left[\frac{V^2_{отр}}{2gL_р} + \frac{1}{3}\left(\frac{1}{K_{взл}} + 2f_{взл}\right)\right] \text{N}, \qquad (1)$$

where G_{to} — is the airplane takeoff weight, N;

V_{uns} — is the airplane unstick speed, m/sec;

f_{to} — is the coefficient of wheel friction on the runway during the takeoff run;

K_{to} — is the airplane L/D at unstick with flaps in the takeoff position.

In Equation (1)

$$\frac{1}{3}\frac{G_{to}}{K_{to}} = \frac{1}{3} c_{x_{to}} S\rho \frac{V^2_{uns}}{2}$$ — is the average value of the drag during the ground run; $\frac{2}{3} f_{to} G_{to}$ — is the average value of the friction force during the ground run.

The value of the velocity V^2_{uns} in the first approximation can be taken as:

for airplanes with TJE

$$V^2_{отр} = 175\frac{p_0}{c_{y_{max}}};$$

for airplanes with PE and TPE

$$V^2_{отр} = 130\frac{p_0}{c_{y_{max}}},$$

where p_0 — is the takeoff wing loading, N/m^2;

$c_{y_{max}}$ — is the maximal value of the lift coefficient with flaps and slats in the takeoff position.

The value of the friction coefficient f_{to} depends on the runway surface condition. On the average we can use the following values of f_{to}:

Dry concrete runway	0.02
Wet concrete runway	0.012
Dry metal runway	0.028
Short grass cover	0.06
Normal grass cover	0.08
Tall grass cover	0.15
Soft dirt	0.20
Hard dirt	0.04
Soft sandy soil	0.12 - 0.30
Wet sticky soil	0.26 - 0.38

One of the primary requirements imposed on the passenger airplane is the possibility of continuing the takeoff and climb with one engine failed. In order to ensure safety of flight in the continued takeoff and climb with one engine out, the vertical rate of climb must be $V_y \geq 2$ m/sec, and the takeoff angle θ must be greater than the minimal permissible angle 1°30'. These values of V_y and θ make the selection of the number of engines definite.

We know from aerodynamics that the equations of motion of the airplane in a climb can be written as

$$P = Q + G\sin\theta;$$
$$Y = G\cos\theta,$$

where P — is thrust, N;
Q — is drag, N;
G — is weight, N;
θ — is climb angle;
Y — is lift, N.

Since

$$c_{x_{to}} = \frac{c_{y_{to}}}{K_{to}}, \quad \text{then} \quad P = \frac{c_{y_{to}}}{K_{to}} S\rho \frac{V^2}{2} + G \sin\theta.$$

But

$$c_{y_{\text{взл}}} S\rho \frac{V^2}{2} = G\cos\theta,$$

then the engine thrust required for climb will be

$$P = G\left(\frac{\cos\theta}{K_{\text{взл}}} + \sin\theta\right).$$

Since with the failure of one engine the angle $\theta = 1.5°$, we can assume that $\cos\theta = 1$. Therefore the engine thrust required for continuing the takeoff with one engine out is found from the formula

$$P_{\text{потр}} = G\left(\frac{1}{K_{\text{взл}}} + \sin\theta\right).$$

The available thrust of all the engines, selected on the basis of the failure of one engine during takeoff, is: for an airplane

with two engines

$$P_{\text{расп}} = 2G\left(\frac{1}{K_{\text{взл}}} + \sin\theta\right); \qquad (2)$$

with three engines

$$P_{\text{расп}} = \frac{3}{2}G\left(\frac{1}{K_{\text{взл}}} + \sin\theta\right); \qquad (3)$$

with four engines

$$P_{\text{расп}} = \frac{4}{3}G\left(\frac{1}{K_{\text{взл}}} + \sin\theta\right). \qquad (4)$$

We use (1) to find the thrust required for takeoff as a function of the required ground run distance, and we use (2), (3), (4) to find the thrust available for takeoff with one engine out.

Comparing the thrust values given by (1), (2), (3), and (4), we conclude that: the number of engines installed on the airplane must be such that the available thrust obtained from the condition of continued takeoff with one engine out is approximately equal to the thrust required to provide the given ground run distance. Moreover, this thrust must be sufficient to obtain the specified cruising speed at the design flight altitude. For example, for an airplane weighing 780 KN at takeoff, wing area 80 M^2, $c_{y_{to}} = 1.6$, $K_{to} = 10$, taking off from a dry concrete runway for a ground run distance L $L_{gr} = 1800$ m, the takeoff engine thrust required to provide the given ground run is 150 KN. The thrust required to continue the takeoff with one engine out is 99 KN. Then the thrust available determined from the single engine failure conditions is 198 KN for the two-engine airplane, 149 KN for the three-engine, and 132 KN for the four-engine airplane.

It is obviously best to install three engines on this particular airplane, since in this case the airplane can continue the takeoff with one engine out. If two engines are installed on this particular airplane, for takeoff with one engine out the thrust available exceeds the thrust required for takeoff with the specified ground run.

If four engines are installed on the airplane, the design case will be that for takeoff with the specified ground run. In this case the thrust available is greater than the thrust determined from the condition of takeoff with one engine out. The reduction of the ground run distance leads to considerable increase of the thrust to provide the given ground run.

Thus, it is best to install two engines on airplanes which take off from short runways. Having a high thrust-weight ratio (F/G_{to} = 0.28 - 0.33), such airplanes can climb more rapidly after takeoff, which reduces the fuel consumption in this stage of the flight. They can fly at higher altitudes than airplanes with a larger number of engines or they can fly at lower altitudes with the engines throttled, which leads to reduction of the fuel consumption and also to increase of engine service life. Installation of two engines is characteristic of airplanes for short and intermediate routes (Tu-124, Tu-134, An-24).

The installation of three engines is advisable on the airplanes for intermediate and long routes (Tu-154). It is best to use four engines on airplanes for long and intercontinental routes (Tu-114, Il-62), which take off from airfields having long runways (2500 m or more). The thrust/weight ratio of such airplanes amounts to 0.22 - 0.27. A characteristic feature of these airplanes is the higher wing loading and considerably lower specific powerplant weight than for the twin engined airplanes. This makes it possible to increase the relative fuel weight, which is very important in obtaining long flight ranges.

Engine selection for supersonic passenger airplanes is influenced by the wide range of flight conditions, and by powerplant economy and reliability. The maximal engine thrust for the supersonic airplane is determined not by the takeoff conditions, but by the acceleration conditions during transition through the sonic barrier. These airplanes have a thrust available during takeoff (thrust/weight ratio 0.4 or more) which permits takeoff from the same runways as those used by the current subsonic airplanes, even in the case of failure of one engine. In this case the total thrust of the remaining engines will be sufficient to complete the takeoff, climb to a safe altitude, and continue flight. The airplane L/D decreases markedly during flight at supersonic cruising speeds, which also requires a high thrust/weight ratio. Published information on the supersonic passenger airplanes now being constructed indicates that they will be equipped with four or six TJE.

Helicopter powerplants are quite varied and depend on the configuration and layout of the helicopters. Helicopter engine selection is outlined in the technical and economic requirements, but we can consider that it is mandatory that at least two engines be installed on passenger helicopters. Provision must be made for the possibility of continuing flight at the best-range speed without descent with one engine operating. With one engine operating at maximum power, the rate of climb should be no less than 1 m/sec.

Vertical takeoff and landing (VTOL) airplanes have been constructed and further development is proceeding at the present time. For these airplanes, powerplant reliability is the primary factor which determines flight safety in the various flight regimes (takeoff, landing, transition), since the required thrust and control of the VTOL airplane are provided by the powerplant. Failure of one engine is the most unfavorable case for the VTOL airplane. With regard to the principle used in creating the vertical and horizontal thrust, three basic VTOL configurations are known:

vertical and horizontal thrust are created by the same engines (integrated powerplant);

vertical thrust is created by one group of engines and horizontal thrust by another group (composite powerplant);

vertical thrust is created by thrust augmentors, and horizontal thrust is created by cruise engines, which are also used for operation of the thrust augmentors (powerplant with thrust augmentors).

VTOL aircraft of the first type most often use TPE, FJE, and TJE. The number of engines depends on the basic thrust/weight ratio, and the thrust/weight ratio after failure of one engine may vary from three to eight. With increase of the basic thrust/weight ratio, the number of engines decreases; for a constant basic thrust/weight ratio, the decrease of the latter after failure of one engine leads to reduction of the number of engines required.

If an integrated powerplant consists of FJE or TJE, then upon failure of one engine the symmetric engine must be shut down for reasons of flight safety. For such VTOL airplanes the number of engines is twice the number of engines of the VTOL airplane using TPE.

In the VTOL airplanes using a composite powerplant, the number of lift engines is selected on the same basis as for the VTOL airplane of the first type. If the composite powerplant consists of lift and lift-cruise engines, their number depends on the basic thrust/weight ratio, the thrust/weight ratio based on the thrust of the lift or lift-cruise engines, and the thrust/weight ratio when one engine fails.

The number of engines required decreases with increase of the overall thrust/weight ratio. For an overall thrust/weight ratio of 1.2 or more and a thrust/weight ratio of the lift or lift-cruise engines varying from 0.3 to 0.7, the number of these engines required will be one or two, and if it is necessary to shut down the symmetric engine the number will be from two to four.

In the VTOL airplane with a powerplant having thrust augmentation units, flight safety with failure of one of the engines is achieved by choosing the required number of engines. Usually the powerplant of such a VTOL airplane consists of two or three engines if the overall thrust/weight ratio amounts to 1.1 - 1.3, with the thrust/weight ratio of the remaining engines being 0.8 - 0.9. We recall that in case of failure of one of the engines (for two engines and two turbofan assemblies — TFA) the vertical thrust of the TFA decreases by only a factor of 1.6 - 1.65 rather than by a factor of 2.

Engine Placement

The placement of aircraft powerplants is extremely varied and depends on the number and type of engines, their dimensions, the mission of the flight vehicle, its performance: flight speed, range, ceiling, load capacity, and so on; the configuration must satisfy the requirements imposed on the powerplant itself.

Placement of Piston and Turboprop Engines

The piston and turboprop engines create thrust by means of the propeller, which defines to a considerable degree the placement of these engines. Powerplants with a single PE or TPE (for sporting, training, and special-purpose airplanes) are usually located in the nose of the fuselage (Figure 4). This scheme has the following advantages:

good access to the engine, propeller, and accessories located on the engine is provided for maintenance;

the fuselage is freed for disposition of cargo and crew;

the engine has low aerodynamic drag, since it is faired into the nose of the fuselage;

the propeller thrust has minimal effect on the stability and controllability characteristics, and the propeller operates in a region of undisturbed flow, which improves its efficiency considerably.

At the same time this engine installation has the following disadvantages:

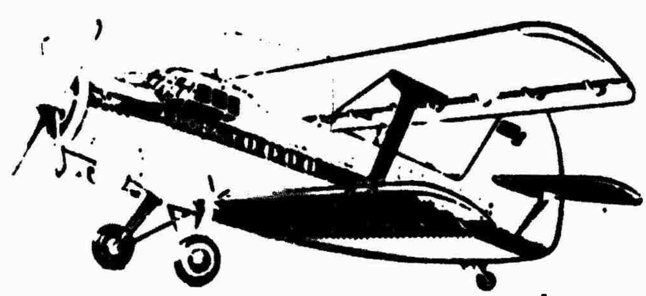


Figure 4. Piston engine installed in fuselage nose

installation of tricycle landing gear with a nosewheel and also the location of other equipment in the fuselage nose, radars for example, is difficult;

the pilot's vision is poor;

the airplane maneuverability is reduced because of the heavy engine being far from the airplane center of gravity;

in a forced landing with the gear up, the propeller is first to contact the ground, which may lead to damage to the engine and the occurrence of fire.

Powerplants with two or four PE or TPE are usually located in the wing leading edge (Figures 5 and 6). The location of the engine nacelles along the wingspan is determined by the propeller dimensions; the powerplant location relative to the fuselage may be determined by the clearance between the prop and the fuselage or by the main landing gear tread.

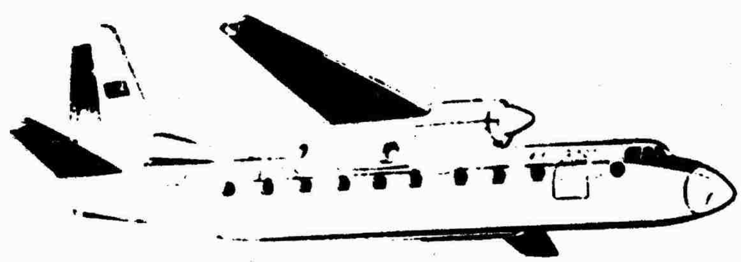


Figure 5. Turboprop engine aligned with upper wing contour

Piston and turboprop engines mounted on the wing are usually located ahead of the wing structure, above the wing structure, or in line with the upper contour of the wing. This engine location scheme has the following advantages:

Figure 6. Turboprop engine above wing structure

the engines relieve the wing load in flight (but during landing and flight in rough air they apply additional loads on the wing) and act as antiflutter balance weights;

full-span trailing edge flaps can be used on the wing;

good access to the engines is provided, engine replacement is convenient, and quick-change powerplant units can be used;

less cabin noise from the engines.

However this arrangement has the following disadvantages:

failure of one of the engines leads to a large turning moment in the horizontal plane and a high rate of roll;

the airplane drag increases markedly because of interference and drag of the nacelles themselves;

in the case of emergency landing with the gear up, damage to the engines (for the low-wing configuration) and fire are possible;

during taxiing dirt, water, and various foreign objects can get into the engines and lead to engine failure.

Turbojet Engine Placement

The absence of propellers on airplanes with TJE opens up broad possibilities for engine placement — from nacelles which are completely outside the wing contour to engines which are entirely buried within the fuselage or wing.

The airplane with one or two TJE usually has the engines in the aft fuselage (Figure 7). This arrangement makes it possible to locate the cockpit in the nose. The propulsive nozzle axis usually coincides with the axis of the aft fuselage and passes through the airplane center of gravity. In this case there is very little airplane trim change when engine power is changed. But the considerable distance between the airplane center of gravity and the engine complicates the balance problem, and the internal resistance of the ducts leading the air to the engine increases (in comparison with airplanes in which the engines are installed in nacelles). It is well known that with increase of the flight speed the internal resistance of the inlet ducting has less effect on decrease of the engine thrust than the external drag of nacelles. Therefore, at high flight speeds it is better to locate the engines inside the flight vehicle, but this is not always possible.

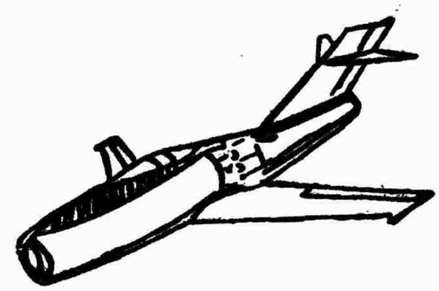


Figure 7. Turbojet engine in aft fuselage

Three engine configurations are used on commercial airplanes with TJE and TFE: in the wing roots, on pylons below the wings, and on the aft fuselage.

When using the first technique (Tu-104, Tu-124), the engines are located aft of the main wing structure (Figure 8) or in the

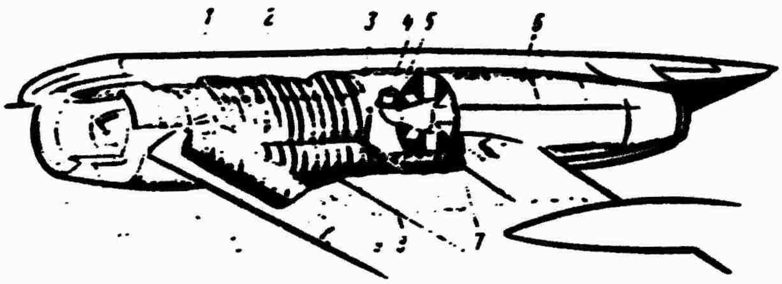


Figure 8. Turbojet engine in wing root behind spars:

1 — wing front spar frame; 2 — upper air duct passing through wing box; 3 — stamped aft spar frame; 4 — turbostarter exhaust duct; 5 — integrated aft section of air duct; 6 — hinged cowl covers; 7 — engine; 8, 10 — wing aft and front spars; 9 — lower air duct; 11 — air intake

main wing structure. Installation of the engines aft of the main wing structure provides good access to the engines and the TJE can be replaced by FJE without major modifications. When the engines are located within the main wing structure (Figure 9), they have shorter air inlet ducting but longer tail pipes to carry the exhaust

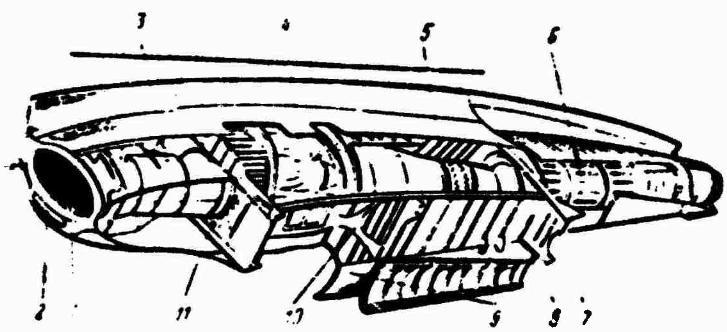


Figure 9. Turbojet engine in wing root between spars:

1, 2 — air intakes of inboard engine and air conditioning system; 3 — air duct; 4 — inboard engine; 5 — firewall; 6 — tailpipe; 7, 11 — rear and front spars; 8, 10 — intermediate and forward firewalls; 9 — hinged hatch covers in lower wing panel for engine installation gases behind the wing trailing edge. In this arrangement the structural weight goes up because of the long tailpipes, and access to the engine is relatively poor.

The arrangement with the engines in the wing roots has the following advantages:

less powerplant aerodynamic drag in comparison with installation of the engines under the wing or on the aft fuselage;

less interference between the fuselage and the wing; with the engines operating an "active" fillet is created, in which the airstream velocity is increased rather than retarded and the pressure decreases in the flow direction. As a result, the overall drag coefficient decreases;

in case of failure of one engine, large yawing and rolling moments do not arise; the relatively slight moments can be counteracted by a small vertical tail area. This makes it possible to replace given engines by engines of higher thrust without major change of the airplane structure;

the safety in an emergency landing is increased, since the engines are protected by the wing against contact with the ground;

the engine arrangement has no effect on wing dihedral, and the latter can be selected solely on the basis of stability and controllability.

The drawbacks of the engine location in the wing roots include:

vibrational loading of the fuselage from the propulsive jet. The closeness of this jet to the fuselage leads to the danger of fatigue failure of the fuselage, and of the empennage as well, which is located in the propulsive jet influence zone;

high noise level in the passenger cabin, particularly where the end of the tailpipe is located;

danger of fire propagation to the passenger cabin and the fuel system if fire occurs in the engine;

possible damage to the pressurized cabin and fuel system if an engine compressor fails;

reduced possibility of wing mechanization because of the presence of the air inlets and exhaust pipes. Design of devices for thrust reversal is difficult, since the jet must be directed forward-down and forward-up. But the portion of the jet which is directed forward-down will be reflected from the airfield surface and into the engine air inlets, and thus affects normal engine operation;

possibility of foreign particle entry into the air inlets from the nose wheels during airplane operation on the airfield;

comparatively difficult conditions for servicing the engine and its accessories.

The arrangement with the engines on pylons below the wings is widely used on foreign airplanes (Figure 10). This scheme has the advantages:

unloading of the wing in flight;

increase of the critical flutter speed because of the forward shift of the center of gravity of those wing sections at which the engines are located;

Figure 10. Turbojet engines below wing on pylons

improved maintenance conditions;

increased fire safety. The use of firewalls in the pylon isolates the burning engine from the wing;

less noise from the operating engines in the passenger cabin;

possibility of noise suppressor and thrust reverser installation on the engines;

use of nacelles with minimal external and internal drag;

absence of propulsive jet effects on the fuselage.

Installation of the engines on pylons has the following disadvantages:

frontal drag is increased because of the presence of the nacelles and pylons, and interference drag shows up where the pylon joins the wing and nacelle;

change of the engine thrust affects the airplane longitudinal and directional stabilities. Engine failure causes a large turning moment in the horizontal plane. High loads are created on the outboard engines during airplane maneuvers;

the landing gear height increases, since the wing dihedral is not selected on the basis of stability and controllability, but rather on the basis that the outboard powerplants not contact the ground in case of a small bank angle occurring during takeoff and landing;

the location of the engines close to the ground increases the probability of engine failure resulting from foreign objects entering the air inlets from the surface of the airfield;

the engine installation on pylons either prevents the use of slats or flaps along the entire wing span or restricts their travel in the area where the engines are installed to less than the design angle;

in case of a forced landing the engines are the first to contact the ground, which may lead to an aircraft fire.

At the present time, extensive use is made of engine installations on the aft fuselage. This scheme is used on the Il-62 (Figure 11), Yak-40, Tu-134, and Tu-154. This arrangement has the following advantages:

an aerodynamically clean wing and effective mechanization along the entire wing span are used to obtain high values of c_y during takeoff and landing;

small yawing moment when one engine fails;

good conditions for air inlet into the engine are created if the engines are far enough away from the fuselage to permit bleeding the boundary layer. When the angle of attack is changed the change of the angle at which the air flow enters the engine inlets changes about half as much as the wing angle of attack changes, while for other arrangements the inflow angle change is greater than the angle of attack change;

the destructive effect of pulsating sonic loads from the propulsive jets on the airplane structure is reduced, since only a small part of the aft fuselage is in the sphere of influence of these pulsations;

the engines are protected against entry of water and foreign objects during takeoff and landing because of the screening provided by the wing and flaps (Figure 12);

the noise and vibration level in the passenger cabin is reduced, improving the comfort level considerably;

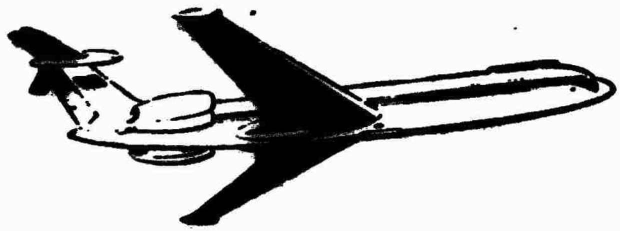


Figure 11. Engine installation on aft fuselage

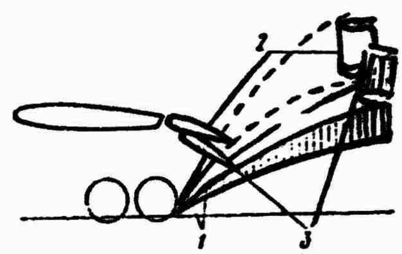


Figure 12. Engine protection against entry of rocks thrown by main gear wheels:

1 — upper boundaries of possible rock trajectory; 2, 3, — positions of flaps and air intakes during takeoff and landing

fire safety is improved because of the increased distance between the engines and the fuel, located in the wing; in case of fire the flame propagates aft without contacting the airplane. In case of a forced landing the engines are protected against contact with the ground by the wing and fuselage;

the longitudinal and directional stabilities are improved because of the fact that the engine pylons and nacelles act as a horizontal tail; the horizontal tail is moved up into a zone of undisturbed flow, and more favorable vertical tail operating conditions are created as a result of location of the horizontal tail on the vertical.

This arrangement has the following disadvantages:

fuselage weight is increased (because of increasing the strength of the aft fuselage), vertical tail weight is increased (since it carries the horizontal tail), and the wing weight is increased (it is not relieved by the engine mass forces);

the airplane center of gravity shifts aft, and therefore the wing must be located closer to the tail. This leads to a longer fuselage nose section, which experiences large bending moments, makes it difficult to provide longitudinal airplane trim, and also complicates longitudinal balancing of the airplane;

the airplane drag is increased owing to the powerplant nacelles and the interference drag, which has a marked effect on the airplane L/D ratio.

With regard to location, helicopter powerplants are divided into two groups: internal, located inside the fuselage, and external, located in separate nacelles outside or above the fuselage. The engine location is determined by the requirement for rational placement of the transmission, possibility of normal engine cooling in all flight regimes and at all times of the year, ease of engine installation and removal, and convenient access to the engine during operation.

On the commercial supersonic airplanes being designed and constructed at the present time the engine nacelles are located below and toward the trailing edge of the wing in order to reduce interference. In this case it is more efficient not to use separate nacelles, but rather a nacelle which is integrated with the airplane structure so that part of the engine case is buried in the wing structure. This nacelle configuration reduces frontal drag, since the pressure is increased around the nacelles, and therefore there is an increase of the lift force without increase of the drag ("positive interference"). The engines can be located in a single nacelle along the airplane centerline or in pairs below the wing. The latter version is preferable from the aerodynamic standpoint, since the engines are not located in the thick fuselage boundary layer. Moreover, the outer portions of the wing are unloaded by the engine mass, while the small increase of the moment of inertia about the airplane longitudinal axis has very little effect.

Engine Mounting

The choice of the engine mount structure depends on the engine type, its installation on the flight vehicle, and also on the magnitude and direction of the applied forces. The following demands are made of the engine mount:

it must accept all the loads which arise under the various flight conditions;

be strong and rigid with minimal weight;

absorb engine and prop vibrations so that they are not transmitted to the flight vehicle structure;

compensate for thermal displacements of the engine case;

provide for rapid installation and removal. At the present time increasing use is being made of quick-change powerplants on flight vehicles. This type of mounting makes it possible to install and remove the engine with all the accessories and components installed on it. In this case the premounted engine can be test run ahead of time.

The engine mount must be designed so that the engine case is not part of the flight vehicle structure. The mount must provide interchangeability of engines, easy and free access to all the accessories which require periodic inspection and adjustment during operation. Maintenance convenience has always been given considerable attention, but today servicing simplicity is considered together with the most important powerplant characteristics, such as reliability, weight, and so on.

Loads Acting

The following forces act on the engine mount during operation: mass, aerodynamic, thrust, and propeller torque. The magnitude of these forces and torque depends on the engine type, its installation, and the maneuvering characteristics of the flight vehicle. The mass forces are determined as a function of the powerplant weight G_{pp}, which includes the weight of the engine and equipment, mount, cowling nacelle, prop, and other components located on the engine.

For airplanes with TPE

$$G_{с.у}=(1{,}9\div 2{,}2)\,G_{дв};$$

for airplanes with TJE

$$G_{с.у}=(1{,}2\div 1{,}6)\,G_{дв},$$

where G_{eng} — is the engine weight.

The mass forces are applied at the powerplant center of gravity and may be directed along the y and z axes (Figure 13). When the airplane rotates about the y or z axis, the gyroscopic moment M_{gyro} arises

$$M_{гир}=I_p\,\omega_x\,\omega_i\,\sin(\omega_x\,\omega_i)\ \text{N}\cdot\text{м},$$

where I_p — is the polar moment of inertia of the prop and rotating masses of the engine, N·m·sec^2;

ω_x* — is the angular rate of rotation of the engine parts, 1/sec

ω_i — is the angular rate of rotation of the flight vehicle about the i^{th} (y or z) axis, 1/sec.

The gyroscopic moment tends to turn the engine axis toward the i^{th} axis, so that when w_x and w_i coincide the rotation of the flight vehicle and engine will be in the same direction.

*Translator's Note: Given incorrectly as "w" in foreign text.

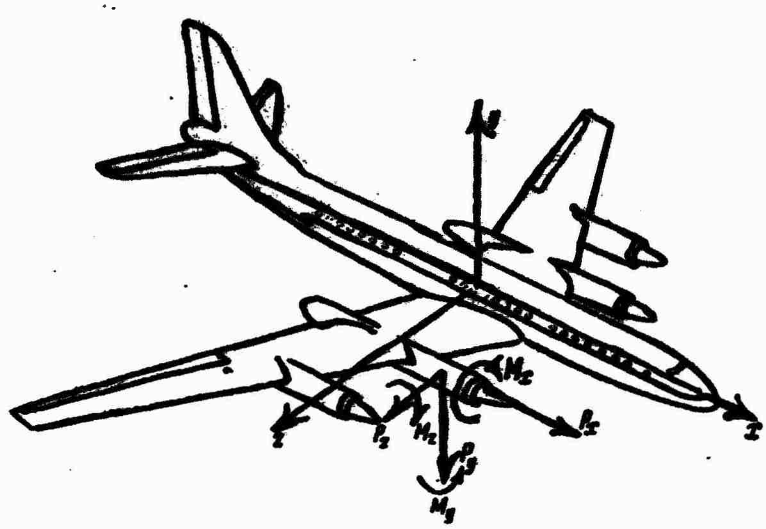


Figure 13. Forces and moments acting on powerplant

The angular velocity of the flight vehicle can be found from the formula

$$\omega_i = 8{,}45\sqrt{\frac{n^{2}_{max}}{p}},$$

where n^{op}_{max} — is the load factor of the corresponding loading case;

p — is the wing loading, N/m^2.

The engine mounts of the PE and TPE are subjected to reactive moment M, directed opposite the prop direction of rotation

$$M = \frac{1}{2\pi}\cdot\frac{N}{n}f\ \text{N}\cdot\text{м},$$

where N — is the power developed on the prop shaft, W;

f — is the safety factor;

n — is the prop rotational speed, 1/sec.

The aerodynamic forces acting on the engine cowlings and nacelles are determined from wind tunnel tests. They are taken into account primarily in the design of the cowlings, nacelles, and the air ducts.

The forces and moments acting on the engine mount can take various different values during airplane operation. The strength standards select several airplane attitudes which result in the most severe cases of loading of the principal parts and components of the airplane. These attitudes are termed the loading cases. The following loading cases are prescribed for the engine mount.

<u>Case</u> A_e — is curvilinear flight of the vehicle at angles of attack corresponding to $c_{y\ max}$. This case occurs in airplane pull-out from a glide, during flight in gusty air, or during entry into a zoom and specifies weight and gyroscopic moment loads on the mounting system.

The operational load P^{op} is

$$P^{э}=\left(n^{э}_{\max_A}+1,5\right)G_{д}$$

and is directed downward perpendicular to the engine axis.

The design load P_{des} is found from the formula

$$P_{расч}=P^{э}f=\left(n^{э}_{\max_A}+1,5\right)G_{д}f.$$

<u>Case</u> A'_e — is curvilinear airplane flight at maximal load factor and maximal dynamic pressure $q_{max\ max}$. This case corresponds to initiation of recovery from a dive with the speed $V_{max\ max}$. The design loading from the weight force and the gyroscopic moment are found similarly to case A_e. However, in addition, consideration is also given to the action of the aerodynamic force P^{op}_{aero} on the cowling, applied downward at a distance 0.25 times the cowling length from its nose

$$P^{э}_{аэр}=c_{y_{кап}}S'_{кап}q_{\max\max},$$

where $c_{y_{cowl}}$ — is a coefficient equal to 0.085;

S'_{cowl} — is the cowling plan-view area.

For this case, the safety factor f = 1.5.

Case D_e — corresponds to the minimal value of the lift coefficient. The operational load P^{op} is applied at the powerplant center of gravity and is directed upward perpendicular to the x axis.

$$P^{э} = \left(n^{э}_{max_D} + 1{,}5\right) G_{д}.$$

The safety factor f = 1.5.

Case D'_e — is curvilinear flight with a negative load factor and the dynamic pressure $q_{max\ max}$. The load for this case is determined similarly to case D_e, but the force P^{op}_{aero} acting on the cowling is directed downward and $c_{y_{cowl}}$ = 0.045.

Case H_e — corresponds to side loading by the mass force. On the mount there act the forces $P^{op}_y = D_e$, $P^{op}_z = \pm 1.5\ G_e$ and the gyroscopic moment.

The safety factor f = 2.

Case M_e (or T_e for the TJE) is engine static operation. The engine mount is loaded by the maximal thrust, the weight G_e, and the prop reactive moment.

The safety factor for this case f = 2.

If the engines have thrust reversal, the calculation is made for maximal reverse thrust. Moreover, the strength norms provide for loading of the engine mount by forces for the combined loading cases

$A_e + M_e (A_e + T_e)$, $D_e + M_e (D_e + T_e)$, $H_e + M_e (H_e + T_e)$, and also by the forces for all the landing gear loading cases with the load factor and safety factors corresponding to the gear design cases. For the combined loading cases, the operational loads are taken in accordance with cases A_e, D_e, H_e, and the thrust and moment values for case M_e (T_e) are taken from the aerodynamic calculation in accordance with the design case examined above.

Configurations

The engine type and its installation on the flight vehicle have considerable influence on the engine mount configuration and structure. Most frequently the engine mount configurations are space frames which connect the engine with the flight vehicle structure and have no less than six members. The members must be arranged to ensure geometric rigidity of the system. However, trusses which have more than six members are frequently encountered; the increased number of members leads to increased survivability of the system. All the engine mount structural elements are made from high-strength alloy steels, heat treated to $\sigma_{ult} = 1100 - 1200$ MN/m^2.

The radial PE mount (Figure 14) consists of a tubular ring to which the engine case is attached and eight members welded to the ring and attached to the airplane structure. The ring and members are made from 30 CrMnSiN steel and are welded together at the joints. The tubes are joined with the aid of gusset plates to strengthen the welded joints. Ears or fittings are welded into the tubes at the places where the truss attaches to the airplane structure. The engine case is attached to the ring with the aid of pins in rubber bushings. The fittings which attach the truss to the airplane structure have rubber dampers.

From the viewpoint of structural mechanics, the engine mount structure is a space truss and falls in the class of complex,

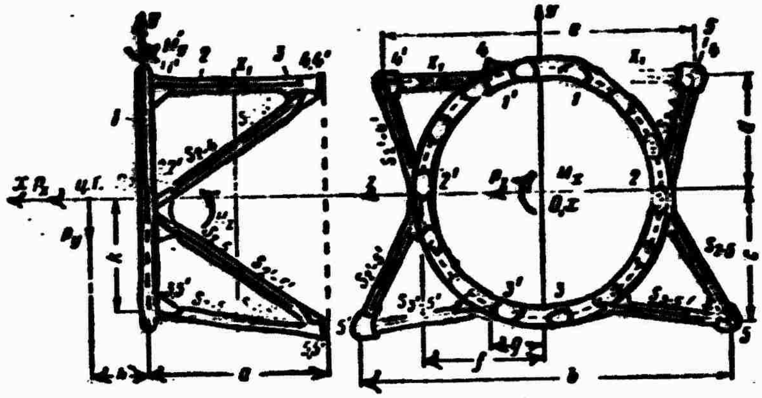


Figure 14. Structure and computational scheme of radial piston engine frame:
1 — frame ring; 2 — brace; 3 — gusset plate; 4 — sockets; 5 — conical support

statically indeterminate systems. The mount joints are welded and can be considered rigid. In this case the tubes will operate in tension-compression and bending when the truss is loaded. However, since the bending is found to be small, in calculations the welded rigid joints can be replaced by ideal hinges operating without friction. The tubular ring and the engine case are assumed to be absolutely rigid bodies. Then the engine mount members will work only in tension and compression.

Mounting of the TPE on the flight vehicle is accomplished with the aid of three-dimensional framework systems which are connected with the engine fittings. The mount may be either the truss or the truss-beam type.

Figure 15 shows the mounting of an engine to the wing center-section by means of a quick-change, three-dimensional, double-tier truss and dampers for attaching the engine to the frame. The engine is attached to the four frame dampers by two forward and two aft trunnions. The front dampers take the prop thrust load and part of the engine weight load, distributed on the basis of the lever rule. The aft dampers take only part of the engine weight.

Figure 15. Truss turboprop mount construction:
1 — engine; 2, 3, 8 — side, upper, and lower braces; 4 — brace-damper; 5 — centerwing spar; 6 — structural truss; 7 — firewall

The engine mount consists of a frame and a structural truss. The frame consists of eight braces: six are made from steel tubes, to the ends of which the attach fittings are welded; the two aft braces are hollow rods with dampers mounted on them. The upper, lower, and aft braces have at one end forks with threaded ends, which are used to align the engine.

The structural truss serves to attach the engine frame and heavy bulkhead to the centerwing spar. It consists of eight braces with fittings for attaching it to the centerwing and fittings for attaching the heavy bulkhead and the frame.

The truss-beam type of engine mount structure (Figure 16) consists of two beams 5 and six braces. The beams can work in bending from the side forces. The braces take only axial loads.

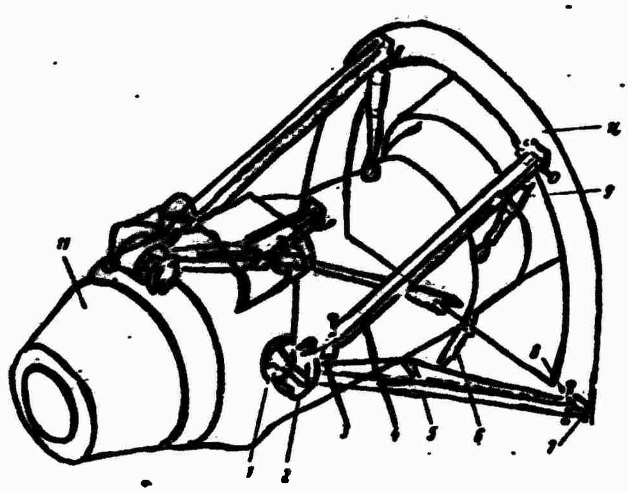


Figure 16: Truss-beam turboprop mount construction:
1 — front trunnion; 2 — front damper; 3 — link; 4 — upper brace, 5 — beam; 6 — inner brace, 7 — bracket 8 — bonding jumper; 9 — brace-damper; 10 — nacelle bulkhead; 11 — engine

The engine is mounted on four trunnions. The front trunnions 1 are inserted into dampers. With the aid of the beams and upper braces the front trunnions transmit the loads to the engine nacelle heavy bulkhead. The load from the aft trunnions is transmitted to the nacelle heavy bulkhead by brace-dampers. The engine position can be altered by adjusting the length of the inner braces 6 and the aft dampers.

The mounting of the TJE on the flight vehicle can be accomplished with the aid of tubular frames, or — if the engine is mounted on an underwing pylon or on the aft fuselage — with the aid of forked fittings located on top of the engine case. Characteristic of the TJE mounting is the presence of heavy rings on the engine itself. The minimal number of heavy rings is two, one of which is primary. On the primary structural ring are located the fittings which take the loads P_x, P_y, and P_z, and also the moments M_x and M_y; on the secondary rings there are fittings which take the load P_y and the moment M_z. The TJE operates in a severe thermal regime, and as a result its case is subjected to considerable thermal expansion. Therefore the engine must be mounted so that case thermal displacements are provided for.

Figure 17 shows TJE attachment to heavy fuselage bulkheads. Characteristic of this arrangement is its asymmetry. Within the bar system consisting of the six primary bars 3, 4, 5 and 7, 8, 9 and the single secondary bar 6, there is the plane element formed by bars 1 and 2.

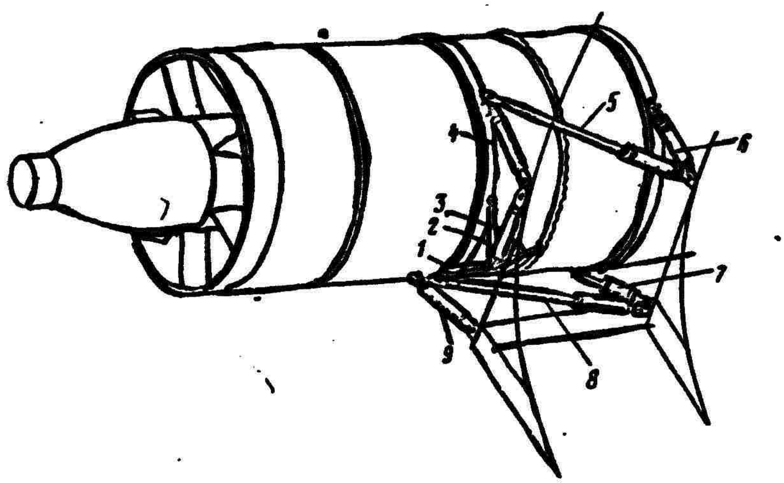


Figure 17. Mount for TJE located in wing root

The correct position of the engine on the airplane is achieved by adjusting the lengths of the six primary bars. Each bar has a rubber damper to absorb vibrations during engine operation.

Of particular interest is the mounting of TJE located on pylons of the aft fuselage (Figure 18). The pylon is an intermediate structural element between the engine and the fuselage. Each engine is

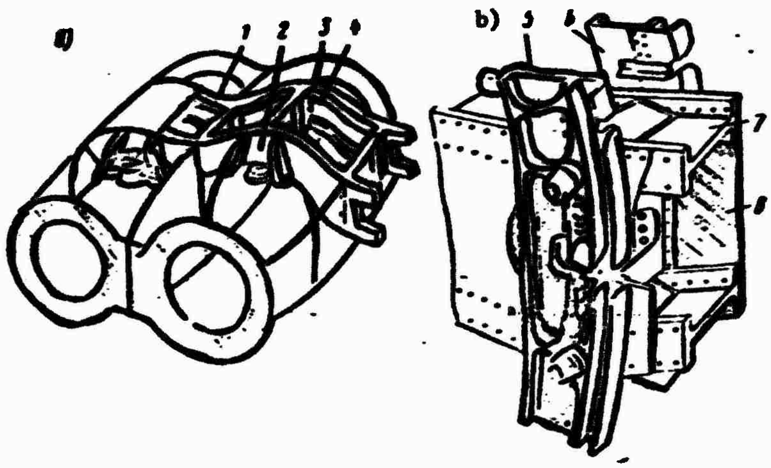


Figure 18. Mount for TJE on aft fuselage pylon:
a — pylon; b — joint between transverse beam and main bulkheads;
1, 3 — cantilever beams; 2, 4 — front and rear mounts; 5, 6, — fuselage bulkheads; 7 — spar; 8 — web

attached at three points to the stamped and machined cantilever beams 1 and 3, two which are attached on each side of the fuselage in the plane of the forward and aft engine mount rings. The cantilever I-section beams, made from high-strength steel, form a box of relatively small height. The upper and lower skins of this box are made from steel and reinforced by stringers. In the aft part of the fuselage there are two heavy machined bulkheads 5 and 6, fabricated from a light alloy, to which transverse horizontal beams having the steel spars 7 and aluminum alloy webs 8 are attached. Fittings which take the loads P_x, P_y, and P_z, and also the moments M_x, M_y, and M_z, are installed on the front beam. The engine attach fittings on the aft beam take the forces P_y, P_z, the moments M_y, M_z, and provide for thermal expansion of the engine. The mounting of TJE on pylons below the wing is accomplished similarly.

The designs of helicopter engine mounting systems are similar to the airplane engine mounting systems. On the helicopter, provision must be made for adjusting the engine mount to align the engine shaft with the transmission.

Strength Analysis

The engine mount must be analyzed for all the cases specified by the strength norms, for which we must determine the loads corresponding to the loading cases.

The radial PE frame (see Figure 14) is doubly statically indeterminate, since it has two redundant members. In view of the fact that it has a plane of symmetry, the analysis can be simplified: all the loads are broken down into a group of symmetric loads and a group of antisymmetric loads. The first include the loads which are identical in magnitude and symmetrically located relative to the plane of symmetry of the structure, and also the loads lying in this plane. The second group includes the loads which are identical in magnitude but are so located on the two sides of the structural plane of symmetry that they become symmetric if the load direction is reversed on one side of the plane of symmetry. In the symmetric members, the stresses from the symmetric loads are equal in magnitude and have the same sign, while in the antisymmetric members the stresses are equal in magnitude and opposite in sign. On the basis of this discussion, the frame shown in Figure 14 becomes a singly statically indeterminate structure.

In determining the stresses in the frame members for the different loading cases, it is convenient to carry out the analysis for the symmetric and antisymmetric loads in general form. We first take into account the action of the symmetric loads and then the antisymmetric.

In making the calculation for the action of both symmetric and antisymmetric loads, we must:

1) select the primary system;

2) determine the stresses S_p which arise in the members of the primary system from the action of the given load;

3) find the stresses S_1 which arise in the members of the primary system from unit forces applied along the direction of the redundant unknown loads in the cut members;

4) calculate the displacements in the primary system from the formulas

$$\delta_{11}=\sum\frac{S_1^2 l}{EF};$$

$$\Delta_{1p}=\sum\frac{S_p S_1 l}{EF},$$

where δ_{11} — is the displacement in the x_1 direction from X = 1;

l, E, F — are the length, material modulus of elasticity, and cross section area of the member;

Δ_{1p} — is the displacement in the x_1 direction from the given load;

5) solve the canonical equation

$$\delta_{11}S_1+\Delta_{1p}=0;$$

$$S_1=-\frac{\Delta_{1p}}{\delta_{11}};$$

6) find the total stress in the members by superposition

$$S=S_p+S_1X_1.$$

Prior to this we must calculate the cosines of the angles formed by the rod directions and the coordinate axes:

Rod	Length of rod projection on axis, m			Rod length, m	Value of cosine		
	l_x	l_y	l_z	$l=\sqrt{l_x^2+l_y^2+l_z^2}$	$\cos(l,x)=\frac{l_x}{l}$	$\cos(l,y)=\frac{l_y}{l}$	$\cos(l,z)=\frac{l_z}{l}$

The analysis of the action of symmetric loads is performed as follows. We select the primary system. To do this we mentally cut the bars 1 - 4 and 1' - 4'. Since the stresses in these bars are identical in magnitude and sign, we denote them by S_1 and direct them away from the engine, considering the bars to be in tension.

We find the stresses S_p in the remaining six bars of the primary system from the given load, assuming the bars to be in tension and considering that the stresses are equal in symmetric bars. Writing the equilibrium conditions, we have

$$\sum M_z = -2S_{p(3-5)}\cos(3-5,x)h - P_y h = 0; \qquad (5)$$

$$\sum M_{3-5} = 2[S_{p(2-4)}\cos(2-4,x)c + S_{p(2-4)}\cos(2-4,y)a] - P_y(a+h) - P_x c = 0; \qquad (6)$$

$$\sum X = -2[S_{p(2-4)}\cos(2-4,x) + S_{p(2-5)}\cos(2-5,x) + S_{p(3-5)}\cos(3-5,x)] + P_x = 0. \qquad (7)$$

From (5) we find the stress $S_{p(3-5)}$

$$S_{p(3-5)} = -\frac{P_y h}{2h\cos(3-5,x)}.$$

We find the stress $S_{p(2-4)}$ from (6)

$$S_{p(2-4)} = \frac{P_y(a+h) + P_x c}{2[\cos(2-4,x)c + \cos(2-4,y)a]}.$$

We find the last unknown stress $S_{p(2-5)}$ from (7)

$$S_{p(2-5)}=\frac{0.5P_x-S_{p(3-5)}\cos(3-5,x)-S_{p(2-4)}\cos(2-4,x)}{\cos(2-5,x)}.$$

After this we must check the correctness of the problem solution. If the stresses in the bars are determined correctly, the sum of the projections of all the forces on the y axis must be zero.

In loading the primary system by the unit forces $s_1 = 1$, we assume the external forces P_x and P_y are zero. Then the equilibrium equations take the form

$$\sum M_z=-2S_{1(3-5)}\cos(3-5,x)\kappa+2\cdot1\cdot\cos(1-4,x)d=0; \qquad (8)$$

$$\begin{aligned}\sum M_{5-5}=2[S_{1(2-4)}\cos(2-4,x)c+S_{1(2-4)}\cos(2-4,y)a]+\\+2\cdot1[\cos(1-4,x)(c+d)+\cos(1-4,y)a]=0;\end{aligned} \qquad (9)$$

$$\begin{aligned}\sum X=2[S_{1(2-4)}\cos(2-4,x)+S_{1(2-5)}\cos(2-5,x)+\\+S_{1(3-5)}\cos(3-5,x)+1\cdot\cos(1-4,x)]=0.\end{aligned} \qquad (10)$$

Solving these equations, we find

$$S_{1(3-5)}=\frac{d\cos(1-4,x)}{\cos(3-5,x)\kappa};$$

$$S_{1(2-4)}=-\frac{(c+d)\cos(1-4,x)+a\cos(1-4,y)}{c\cos(2-4,x)+a\cos(2-4,y)};$$

$$S_{1(2-5)}=-\frac{S_{1(2-4)}\cos(2-4,x)+S_{1(3-5)}\cos(3-5,x)-\cos(1-4,x)}{\cos(2-5,x)}.$$

The correctness of the solution is checked by vanishing of the sum of the projections of all the forces on the y axis. We summarize the results in a table which is used to calculate the coefficients δ_{11} and Δ_{1p}, and also the actual stresses which arise in the members from the symmetric loading

Rod	l, m	$S_p \cdot N$	$S_1 \cdot N$	$S_1^2 l$	$S_p S_1 l$	$S_1 X_1$	$S_{sym}=S_p+S_1X_1$

The next step is to analyze the mount subject to the antisymmetric loads P_z, M_x, M_y, M_z.

We take the same primary system as before. As before, we mentally sever the forward part of the powerplant and write the equilibrium conditions for the severed part. We consider the rods to the left of the plane of symmetry to be in tension, those to the right — in compression. Satisfying the equilibrium conditions ($\Sigma Z = 0$, $\Sigma M_x = 0$, $\Sigma M_y = 0$) for the system, we obtain three equations with the three unknown force factors $S_{p(2-4)}$, $S_{p(2-5)}$, $S_{p(3-5)}$. The solution of these equations yields the expressions for determining the unknown stresses $S_{p(2-4)}$, $S_{p(2-5)}$, $S_{p(3-5)}$ (the technique for formulating the equations and their solution is similar to that presented earlier in examining the action of symmetric loads).

Then we load the primary system with the forces $X_1 = 1$ and calculate the stresses in the rods, assuming the external loads to be zero. Satisfying the equilibrium conditions, we find $S_{1(2-4)}$, $S_{1(2-5)}$, $S_{1(3-5)}$. We formulate an auxiliary table which we then use to calculate the stresses in the rods

Rod	l, м	s_p, N	s_1, N	$s_1^2 l$	$s_p s_1 l$	$s_1 x_1$	$S_{antisym} = S_p + S_1 X_1$

We find the overall stresses in the rods when any arbitrary loads act on the mount as the sum of the two stresses from the action of the symmetric and antisymmetric loads

$$S = S_{\text{с.им}} + S_{\text{ос. сим}}. \quad (11)$$

We note that (11) can be used only in those cases in which the stresses in the rods do not exceed the proportional limit, and none of the compressed rods buckles. We assume that these conditions are satisfied under the action of the operational loads.

If individual rods begin to operate in the plastic deformation region or the compressed rods buckle, the action of each such rod on the mounting system is replaced by the force S_1, equal to the stress in the rod at which yielding of the material begins or the critical force (in compression). In these cases the stress S_1 is considered a given external load, and the further calculation is made using the method described above.

<u>The analysis of the truss-type TPE mount</u> (see Figure 15) is made similarly.

<u>The analysis of the truss-beam type TPE mount</u> (see Figure 16) can be made under the assumption that the beam 5 from the engine trunnion to the attachment to the inner brace 6 is a rod. Since the rod 6 is not connected directly with the engine, the engine is mounted by six rods and the system is statically determinate. Writing the equation for the moments about the front engine mount trunnion, and also the sum of the projections of all the forces on the x and y axes, and equating the projections to zero, we can determine the stresses in the rods and beam. The resulting stress in the beam at the point where it joins the inner brace 6 is resolved along two directions — along the inner brace 6 and along the beam from the brace attach point to the nacelle bulkhead. After this, these stresses are easily found. The beam works in bending under the action of the force P_z.

<u>The determination of the stresses in the members of the asymmetric TJE mount truss</u> (Figure 19) is somewhat more complex, since this requires the solution of a larger number of equations. The

minimal number m of rods of a space frame with plane elements is defined by the equation

$$m=6+3n'+2n'',$$

where n' and n" are the numbers of space and plane elements respectively.

In the truss in question n' = 0 and n" = 1. Therefore, the minimal number of rods must be eight. In reality the truss has nine rods. Let us cut the truss so that rods 2 - 2' and 2 - 2" remain with the engine; then the truss will have a single redundant rod. We can take any of the rods as the redundant member. If we assume that rod 11 - 12 is redundant, we can find the stresses in the remaining six rods (1 - 8, 3 - 2, 4 - 5, 10 - 6, 9 - 7, 9 - 13) of the primary system resulting from the external loads. To do this we must write and solve six equilibrium equations. Then we find the stresses in the rods from unit load acting in the redundant rod, and after this we find the overall stresses in the rods. Knowing the stress in rod 2 - 3, we can find the stresses in rods 2 - 2' and 2 - 2" by examining equilibrium of the plane element 2.

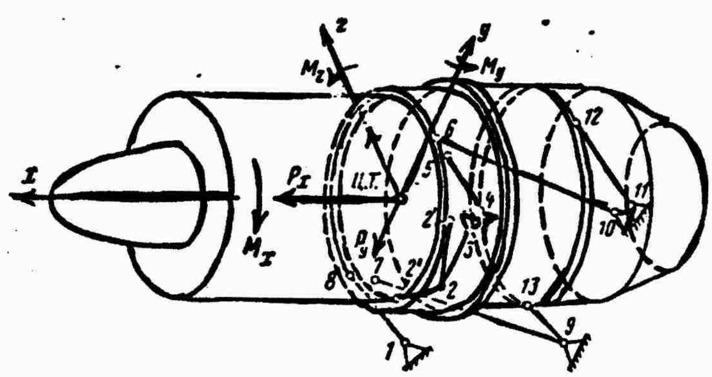


Figure 19. Computational scheme for asymmetric TJE mount

Powerplant Vibration

The engine and propeller are vibration excitation sources. The vibration amplitude in certain cases reaches a magnitude such that it interferes with normal operation of the instruments and mechanisms, and failure of structural elements may occur. Vibrations reduce the service life and reliability of components and equipment, and may give rise to fatigue failure of the structure. The engine creates vibrations of two types: mechanical and sonic. The mechanical vibrations are created by torque unbalance, unbalanced inertial forces of the translating components, moments of these forces, and dynamic unbalance of the rotating engine components. The propeller gives rise to similar mechanical and sonic vibrations. The mechanical impulses are caused by incomplete propeller static, dynamic, and aerodynamic balance.

Propeller static unbalance occurs when the propeller center of gravity does not coincide with its axis of rotation. The magnitude of these impulses is determined by the propeller unbalance tolerance.

Propeller dynamic unbalance occurs when the centers of gravity of the individual blades do not lie in the propeller plane of rotation.

Aerodynamic unbalance occurs as a consequence of the fact that the blades have different incidence angles and this means that they have different values of the thrust, or when the blade centers of pressure lie at a different distance from the axis of rotation. Aerodynamic unbalance has a large magnitude on variable-pitch propellers when the blade rotation mechanism operates improperly. Vibrations resulting from propeller gyroscopic effects arise during propeller operation in flight. These vibrations develop during curvilinear motion of an airplane with two-bladed propellers, because of inequality of the moments of inertia of the propeller about the principal axes. The influence of the wing on the propeller also has an effect.

The disturbing forces which arise during operation of the powerplant are periodic, multiples of the rpm of the rotating engine components and propeller. The structure executes forced vibrations under the influence of these forces.

Since several disturbing periodic forces usually act on the mount, the displacements caused by these forces are the result of superposing the displacements caused by each force individually. Thus the relationship between the natural vibration frequency p and the disturbing force frequency ν is important.

The engine (propeller) vibration frequencies ν depend linearly on the engine rotor (propeller) rpm. The order of the harmonics of the engine disturbing forces and moments which cause vibrations are:

for PE: $\frac{1}{2}$, 1, 2, $\frac{a}{2}$, $\frac{a+1}{2}$ (a is the number of cylinders);

for TJE and TPE: 1, 2, ... The order of the propeller harmonics for a k-bladed propeller are: 1, 2, ..., k, mk (where m = 2, 3, ...). The harmonic order is defined in relation to the rotational speed of the crankshaft (for the PE), turbine (for the TJE and TPE), and propeller.

Most critical are the vibrations with frequencies n and 2n for airplanes with PE; n for airplanes with TJE; n_p and $2n_p$ for airplanes with TPE (n and n_p are the rotational speeds of the engine and the propeller.

The problem of vibrations caused by the aerodynamic action of the propeller on the fuselage has become a critical area for modern airplanes with TPE. These vibrations have comparatively high frequency and can lead to fatigue failure of various fuselage structural elements, which is particularly hazardous for airplanes with pressurized cabins.

Usually the engine mount natural vibration frequency lies in the frequency range of the exciting vibrations, so that resonance can occur. In themselves the engine exciting forces are of small magnitude; however, a large dynamic effect is created at resonance.

To obtain minimal stresses in the engine mount and to avoid the resonance zone, it is necessary that the powerplant natural vibration frequency be lower than that of the exciting forces.

The engine with its accessories has six degrees of freedom: translational displacement in the direction of the x, y, z axes and angular displacement about these axes. Depending on the location of the powerplant center of gravity and the engine mount center of stiffness, the natural vibrations may be uncoupled, doubly-coupled, and triply-coupled. Uncoupled vibrations arise when the powerplant center of gravity coincides with the engine mount center of gravity. Usually the vibrations are triply-coupled, and displacement along any axis causes displacement and rotation relative to the other two axes.

The powerplant natural vibration frequency for vibration along the axis is

$$p=\sqrt{\frac{K_s}{M}},$$

where K_s — is the stiffness of the entire suspension in the direction of the sought displacement;

M — is the powerplant mass.

The force which produces unit deformation in the direction of its action is called the stiffness.

The powerplant natural vibration frequency for vibration about the axis is

$$p'=\sqrt{\frac{R}{I}}.$$

where R — is the stiffness of the entire suspension about the axis of the sought vibration;

I — is the moment of inertia of the engine rotor relative to the sought vibration.

Powerplant stiffness is different in different directions and depends on the stiffnesses of the engine mount and the dampers; it is found from the formula

$$K_{\text{с.у}}=\frac{K_{\text{м}}K_{\text{а}}}{K_{\text{м}}+K_{\text{а}}}.$$

where K_{pp} — is the powerplant stiffness in one direction;

K_m and K_d — are the stiffnesses of the mount and the vibration damper in the same direction.

The shock absorption stiffness depends on the type of dampers used and their arrangement relative to the powerplant center of gravity.

Utilizing the resonance phenomenon, we can with the aid of a vibrator, which is a dynamically unbalanced system mounted on the engine, determine the powerplant natural vibration frequency experimentally. As the vibrator rotates, unbalanced inertial forces develop in the vibrator and are transmitted to the powerplant. Under the influence of these vibrations, the powerplant begins to vibrate with a frequency equal to that of the vibrator. The vibrations are recorded on an oscillograph. On the basis of analysis of the vibration regimes, we can find the exciting harmonics, their order, the source, and the resonant regimes. The vibration frequency at the moment of resonance will then be the powerplant natural vibration frequency.

Engine Mount Damping

Engine mount damping is provided to reduce the harmful effect of powerplant vibrations created by engine operation on the flight vehicle structure, and to reduce the forces transmitted to the supports by reducing the powerplant natural vibration frequency. Even in those cases where there is no resonance, damping is used to reduce the fatigue loads in the mount and also the unpleasant sensations felt by the passengers and crew owing to the vibrations.

The basic technique for improving the damping is to increase the ratio ν/p. The essence of damping is that the engine connection with the flight vehicle is not rigid, but rather by means of flexible elastic elements, which reduce markedly the powerplant vibration frequency.

Dampers can be installed both in the fittings which attach the engine to the mount and in the fittings which connect the mount with the airframe. The latter location of the dampers is not desirable, since it leaves the mount in the sphere of influence of the exciting impulses. In this connection the vibrational loads are increased and there is more danger of resonance for the mount. Moreover, there is an increase of the distance between the dampers and the engine center of gravity, as a result of which the dampers will experience large static loads.

Damper-braces are sometimes introduced into the mount design to provide damping. In some cases dampers are installed both at the fittings attaching the engine to the mount structure and at the fittings where the mount is attached to the airframe. This type of damping isolates the airframe from the engine vibrations most completely.

The introduction of damping may create conditions for free vibration of the powerplant (permitting all degrees of freedom).

However, during vibrations about the y and z axes precessional vibrations caused by the gyroscopic effect of the engine rotor and the propeller develop, and these are damaging to the propeller and engine rotor. In this connection the torsional stiffness about these axes must be reduced. The largest exciting impulses act along the y and z axes and about the x axis. Therefore the most effective damping is required in these directions. In the other directions, the permissible stiffness of the damping and of the dampers themselves is several times less (up to 10). Moreover, the mount structure stiffnesses and natural vibration frequencies are different in the different directions, which requires different damping stiffness in the corresponding directions.

Rubber, for which the shear modulus G_{sh} is less by a factor of 6 - 10 than the compression modulus G_{com}, is used as the damper elastic element. By using dampers with different thickness of the rubber in different directions or forcing the rubber to work in shear in one direction and in compression in another direction, we can obtain the required damping stiffness in different directions.

Such dampers, as shown in Figure 20, have different thicknesses of the rubber in different directions. These dampers are attached to the support ring of the radial PE mount. They provide large angular displacements about the x axis, thus damping the torque impulses, permit relatively large displacements along the y and z axes, and small displacements along the x axis. The rubber works in compression in all directions. This damper has the lowest stiffness along the y and z axes and the greatest stiffness in the direction of the x axis.

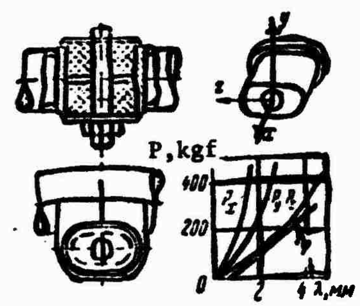


Figure 20. "Sleeve" type damper and its characteristic curves

The damper stiffness is

$$K_a = \frac{dP}{d\lambda} = \mathrm{tg}\,\varphi,$$

where P — is the acting force;

λ — is the displacement.

We see from Figure 20 that the damper stiffness is not constant but depends on the magnitude of the preload and is greater the larger the preload.

Figure 21 shows the dampers of the TPE whose mount is shown in Figures 15 and 16. The dampers are located in the plane parallel to the engine shaft axis. In this case, in order to give the entire system good mobility about the x axis and the highest possible stiffness about the y and z axes, the lateral distance between the dampers

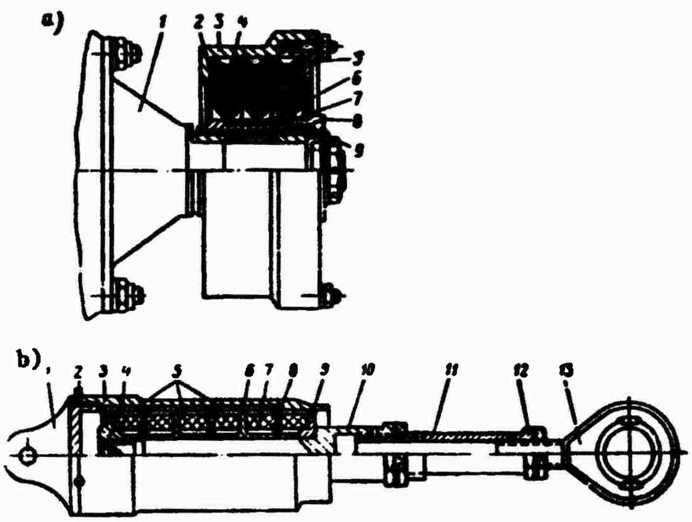


Figure 21. Dampers:

a — front (disk): 1 — engine trunnion; 2 — washer; 3 — body; 4 — damper disk; 5 — flange; 6 — sleeve; 7 — lockwasher; 8 — nut; 9 — sleeve damper

b — aft (brace-damper): 1 — fork; 2 — lockscrew; 3 — body; 4 — nut; 5, 6, 9 — sleeves; 7 — damper disk; 8 — ring; 10 — central rod; 11 — sleeve; 12 — lockwasher; 13 — ear is made as small as possible and the distance between the forward dampers and the brace-damper is made as large as possible.

The forward damper (Figure 21a) consists of three disks 4 and a single sleeve-type damper 9. Each disk consists of three washers 2, between which two layers of rubber are vulcanized. Two washers are restrained in the case 3; the third is seated on the damper. The sleeve-type damper 9 consists of two sleeves between which rubber is vulcanized.

The aft damper (brace-damper) is assembled from three rubber disks 7 (Figure 21b).

In installing the dampers, we must bear in mind that fuel and oil have a damaging effect on the rubber used in the dampers. Therefore the dampers must be enclosed in protective covers or cases. We must also remember that the elastic properties and strength of the dampers decrease with temperature reduction and increase with temperature increase.

Engine Nacelles

In order to reduce external drag and organize the air flow, the engines are enclosed in nacelles, which create a smooth transition from the engine to the airframe and protect the engine against contamination.

The engine nacelle must provide air to the engine with a uniform velocity field for normal engine operation and cooling, minimal external engine drag, convenient access to the engine and the accessories mounted on the engine for inspection, adjustment, and replacement, and in addition should be light in weight.

The engine nacelle consists of the cowling and the body, which form the structure. The cowling configurations may be of the frame or panel type.

In the frame types the strength and stiffness are provided by the frame, to which are attached removable cover panels with thin skin reinforced by stringers.

The panel type cowling consists of rigid panels which are interconnected by tension latches and form a closed structural shell. The load from the cowling is transmitted to the engine and through the engine mount to the airframe. Simultaneous attachment of the cowling to the engine and the airframe is not permitted. The stiff cowl cover panels are hinged rather than removable. In the open position the cover is held by special braces. Sealing strips are provided on the panel edges to provide tighter fit of the cover in the closed position.

Every powerplant has several accessories which require access during operation (oil tank filler necks, drain cocks, and so on). Small hatches with removable or hinged cover plates are provided in the panels for access to these components.

Of all the cowling elements, the highest loads are experienced by the air intake, therefore it is usually a one-piece structure. The engine nacelle has a firewall to isolate the engine hot sections from the sections which are cooler but hazardous from the fire standpoint.

The PE nacelle (Figure 22) consists of the propeller spinner, cowling, and body. The cowling is attached to the engine and consists of the air intake, front ring, and intermediate and aft sections. The nacelle body takes the loads from the engine mount system and transmits them to the airframe.

The TPE nacelle (Figure 23) consists of the propeller spinner, cowling, and body. The propeller spinner serves to reduce the external drag, shape the engine inlet duct, and protect the hub against external damage. The engine cowling consists of the gearbox fairing, air inlet, upper beam, and the lower and two side covers.

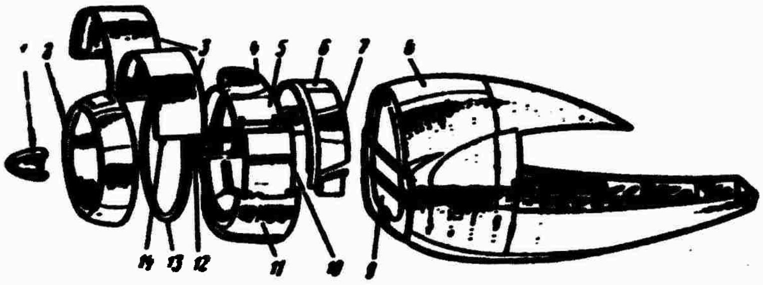


Figure 22. Piston engine nacelle:

1 — prop spinner; 2 — front cowl ring; 3 — intermediate cowl cover; 4 — air intake; 5 — upper aft cowl cover; 6 — skirt; 7 — controllable flaps; 8 — body; 9 — firewall; 10 — removable side cover of aft cowl; 11 — lower cowl cover; 12 — side panel; 13 — intermediate cowl bulkhead; 14 — lock

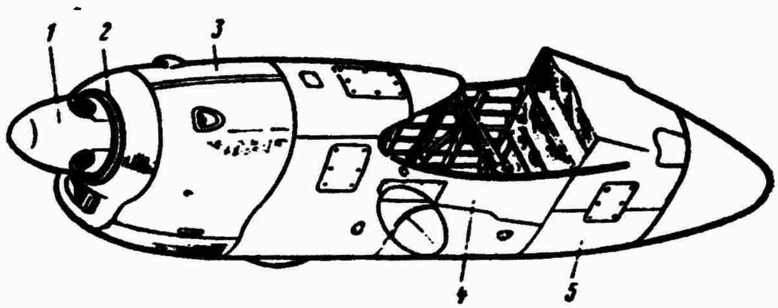


Figure 23. Turboprop nacelle:

1, 2 — prop spinner and gearbox fairing; 3 — cowl; 4 — body; 5 — aft section

The gearbox fairing, which is a continuation of the spinner, forms the inner contour of the engine air inlet duct.

The inner side of the air intake forms the outer contour of the engine inlet duct. The air intake bulkhead together with the skin create the annular chamber of the air intake anti-icing system, which is supplied with hot air from the engine compressor.

The engine nacelle is attached to the wing and forms the structural element of the engine mount.

The nacelle (Figure 24) of a TJE buried in the wing includes the air intake, intermediate section, cowl, and exhaust fairing. The air intake consists of stringers, frames, and skin. The intermediate section of the engine nacelle consists of a body with covers and air ducts. The engine cowl is located aft of the wing rear spar and is a continuation of the nacelle intermediate section. The exhaust fairing forms the smooth aft end of the nacelle and directs the hot gas jet away from the fuselage.

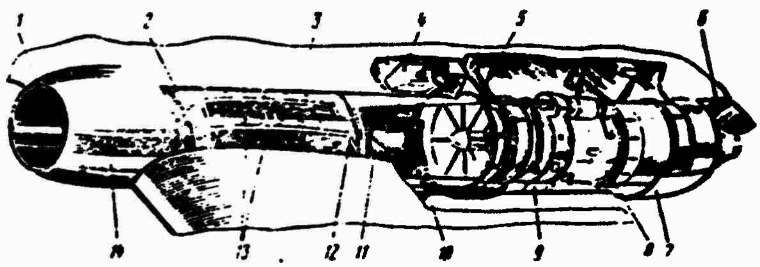


Figure 24. Turbojet nacelle:
1 — air duct; 2, 12 — front and rear spars; 3 — intermediate section; 4 — oil tank; 5 — oil tank vent; 6 — exhaust fairing; 7 — engine cowl; 8 — engine; 9 — oil pump; 10 — drain tube; 11 — fuel-oil radiator; 13 — stringers; 14 — air intake

Engine Mount Maintenance

During engine maintenance an inspection is made of the welded assemblies and safetying of all nuts, checks are made for scratches and damage to paint finish, and absence of corrosion is verified. During major overhaul of the engine mount, magnetic or similar inspection is used to find any cracks or microcracks. The mount must be replaced immediately if cracks are found in the welded assemblies.

Engine replacement is accomplished when the prescribed service life runs out or in case of engine failure. Safety rules must be strictly observed during engine replacement.

Contact of oil, gasoline, or electrical components cannot be permitted. When replacing an engine the pressurization, hydraulic, and fuel system lines are disconnected, electrical plugs, bonding wire, engine control cables, and the tailpipes are disconnected. All tubes and electrical plugs must be covered after being disconnected. The openings in the engine must be plugged.

Prior to removing the engine, the fire shutoff valve is closed, the remaining fuel is drained from the line downstream of the fire shutoff valve. The oil is drained from the powerplant oil system, then internal preservation is performed in accordance with the engine maintenance instructions.

After this the engine is removed. If the powerplant has a propeller, it is removed first.

External preservation of the removed engine must be performed.

The engines being installed must be in good condition, completely outfitted, and must have all the required documentation. Prior to installation, the engine is depreserved and the accessories and components are mounted on it. The alignment of the engine centerline is checked against the leveling diagram.

After the first test run of a newly installed engine, the normal powerplant postflight servicing operations are performed.

Engine alignment is accomplished by means of adjustable length rods. Such rods have adjustable sleeves at one end into which fittings are screwed. Each sleeve has an outer right-hand and an inner left-hand thread. The length can be changed during alignment by turning the sleeve in either direction. In so doing it is necessary to be certain the brace adjusting sleeves cover the special inspection openings and marks.

After installing the engine in the proper position, all the locknuts, dampers, and control rods must be tightened securely and safetyed.

Dampers. The engine mount systems are removed and replaced by new systems when their established service life runs out. The dampers are packed and stored in accordance with the general instructions for rubber parts.

All dampers are subjected to tests for radial and axial loads after a year of shelf storage, and the forces applied must cause the corresponding deformations. The magnitude of the residual deformations must not exceed the specified values.

CHAPTER II

HYDRAULICS OF SYSTEMS

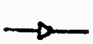
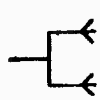
Graphical Representation of Systems

Drawings and their simplified and generalized versions in the form of block, schematic, and installation diagrams are used for graphical representation of systems. Such diagrams are drawn following the rules called for by the Unified Design Documentation System.

The block diagram can be used to determine the basic components of the system, their purpose and interconnection. The components are represented in the form of rectangles within which their names are placed, the connections are represented by lines, and the flow direction is shown by arrows.

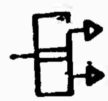
Schematic diagrams show the components and connections between them, using the conventional symbols (Table 1) presented in standard manuals and industrial norms. Sometimes the components are represented in the form of schematic cross sections.

TABLE 1. CONVENTIONAL SYMBOLS ON SYSTEM SCHEMATIC DIAGRAMS

Name	Symbol	Name	Symbol
Pneudraulic accumulator		Transmitter: level control:	
Tank		capacitance	
Bottle		float	
Transmitter:		Restrictor	
pressure		Air intake: from atmosphere	
flow		from engine	
temperature		Filler neck, refueling fitting	
Measuring device:		Differential regulating valve ($p_1 - p_2$ = const)	
pressure: warning			
indicating		Cock	
flow		Heater	
temperature		Flow direction: liquid	
level		air (gas)	
Nozzle manifold			

(Table continued on following page)

TABLE 1 (Continued)

Name	Symbol	Name	Symbol
Valve :		Pump:	
check		general symbol	
regulating:			
normally closed (drain)		gear	
normally open (fill and transfer)		Radiator	
limiting $p_1 < p_{max}$)		Liquid-gas separator	
reducing (p_2 = const)		Hydraulic line	——
		Filter	
		Electric wiring	

The installation diagram is used to determine the actual location of the lines and connections between the components. These diagrams are drawn to scale in either rectangular or perspective projections. The components are represented in the form of simplified geometric contours. If the installation diagram is not drawn to scale, it is a connection diagram.

On colored diagrams the lines and components are colored just as on the actual installation: fuel system — yellow; oil — brown; air — black; fire extinguishing — red.

Hydraulic Resistance

The circuits through which the liquid (gas) travels consist of components, lines, and fittings. They may be divided into simple and complex. We term those circuits which have a line of constant diameter without branches simple circuits, and those having a line of variable diameter or branches complex. We shall examine the hydraulic resistances of the simple and complex circuits.

Simple Circuit

The hydraulic resistance Δp_h to fluid flow through the circuit is found from the formula

$$\Delta p_{\text{г}} = \Delta p_{\text{тр}} + \Delta p_{\text{м}} \text{ N/m}^2, \tag{12}$$

where Δp_{fr} — is the frictional resistance to fluid flow through the line;

Δp_l — are the local resistances.

The frictional resistance is found from the formula

$$\Delta p_{\text{тр}} = \lambda \frac{l}{d} \cdot \frac{\gamma V^2}{2g} \text{ N/m}^2, \tag{13}$$

where λ — is the frictional resistance coefficient;

l — is the line length, m;

d — is the line internal diameter, m;

γ — is the fluid specific weight, N/m^3;

V — is the fluid velocity, m/sec;

g — is the acceleration of gravity, m/sec^2.

For lines whose internal sections have arbitrary shape, we define the hydraulic diameter d_h

$$d_{\text{г}} = \frac{4f}{П} \text{ м}, \tag{14}$$

where f — is the line internal section area, m^2;

Π — is the internal section perimeter, m.

The frictional resistance coefficient λ is calculated as a function of the fluid flow regime, defined by the Reynolds number: $Re = \frac{Vd}{\nu}$, where ν is the kinematic viscosity, m^2/sec.

If the viscosity is expressed in Stokes (cm^2/sec) or in centistokes (mm^2/sec), we must bear in mind that 1 m^2/sec = 10^4 St = 10^6 cSt.

Expressing the velocity V through the flowrate W and the line internal section area f, we obtain

$$Re = \frac{Wd}{f\nu}.$$

For a line with circular internal section

$$f = \frac{\pi d^2}{4}\ m^2.$$

Then

$$V = \frac{W}{f} = \frac{4W}{\pi d^2} \qquad (15)$$

and

$$Re = \frac{4W}{\pi d\nu}. \qquad (16)$$

For the laminar fluid flow regime, when Re < 2300, the frictional resistance coefficient

$$\lambda = \frac{64}{Re}. \qquad (17)$$

For the turbulent fluid flow regime without account for line roughness and heat transfer, for $4 \cdot 10^3 < Re < 10^5$, the frictional resistance coefficient

$$\lambda = \frac{0{,}316}{\sqrt[4]{Re}}. \qquad (18)$$

If $10^5 < Re < 5 \cdot 10^6$, then

$$\lambda = \frac{0{,}09}{\sqrt[7]{Re}}. \qquad (19)$$

When $4 \cdot 10^3 < Re < 5 \cdot 10^6$ we can also use the Konakov formula [34]

$$\lambda = \frac{1}{(1{,}81 \lg Re - 1{,}5)^2}. \qquad (20)$$

For flexible lines (hoses) the frictional resistance coefficient λ_{flex} is larger than for rigid lines, and it is usually assumed that

$$\lambda_{т.ш} = 1{,}3\lambda.$$

The <u>local resistances</u> are calculated from the formula

$$\Delta p_м = \sum \zeta_м \frac{\gamma V^2}{2g} \quad \text{N/m}^2, \qquad (21)$$

where ζ_l — is the local resistance coefficient.

The local resistance coefficients ζ_l presented in Table 2 are approximate values. For many components ζ_l decreases with increase of the fluid velocity (Reynolds number) (Figure 25). Therefore it is more accurate to determine Δp_l from curves (Figure 26).

On the basis of (13) and (21), we can write (12) as

$$\Delta p_r = \lambda \frac{l}{d} \frac{\gamma V^2}{2g} + \sum \zeta_м \frac{\gamma V^2}{2g} = \left(\lambda \frac{l}{d} + \sum \zeta_м\right) \frac{\gamma V^2}{2g} = \zeta_э \frac{\gamma V^2}{2g}, \qquad (22)$$

TABLE 2. LOCAL RESISTANCE COEFFICIENTS OF COMPONENTS AND FITTINGS (REFERRED TO VELOCITY AT THEIR INLET FITTINGS)

Component, fitting	ζ_l
Inlet to tank from line	1.0
Exit from tank to line	0.5
Elbow, rounded (90°)	0.2 - 0.4
Shutoff cock	1.0 - 2.5
Check valve	1.7 - 2.0
Flowmeter	7.0 - 10.0
Flexhose connection	0.2 - 0.3
Self-closing connection	2.0 - 2.5
Tee	1.5 - 2.5
Elbow	1.2 - 1.3
Screen	1.5 - 2.5

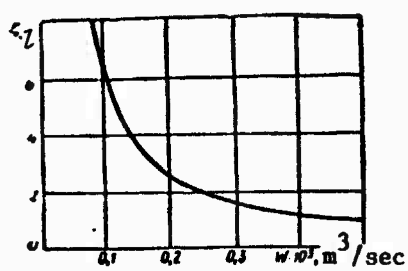


Figure 25. Local resistance coefficient of fuel screen versus flowrate

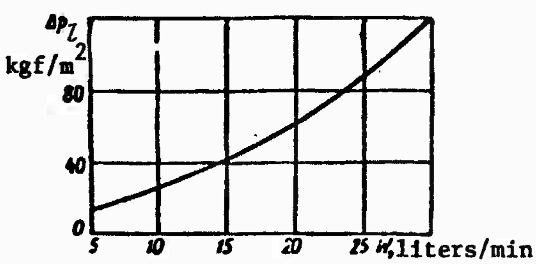


Figure 26. Hydraulic resistance of fuel screen versus flowrate

where ζ_{eq} — is the equivalent hydraulic resistance coefficient for fluid flow through the circuit.

This coefficient equals

$$\zeta_{\text{э}}=\lambda\frac{l}{d}+\Sigma\zeta_{\text{м}}. \tag{23}$$

Substituting (15) into (22), we obtain

$$\Delta p_r=\frac{8}{\pi^2 g}\cdot\frac{\zeta_{\text{э}}}{d^4}\gamma W^2=c^2k^2\gamma W^2, \tag{24}$$

where

$$c^2=\frac{8}{\pi^2 g}=0{,}083\ \text{sec}^2/\text{m};$$

$$k^2=\frac{\zeta_{\text{э}}}{d^4}=\frac{\lambda\frac{l}{d}+\Sigma\zeta_{\text{м}}}{d^4}. \tag{25}$$

The coefficient

$$k=\frac{\sqrt{\zeta_{\text{э}}}}{d^2}=\frac{\sqrt{\zeta_{\text{э}}}}{\frac{4}{\pi}f}=\frac{\sqrt{\lambda\frac{l}{d}+\Sigma\zeta_{\text{м}}}}{d^2}\ 1/\text{м}^2. \tag{26}$$

is called the reduced hydraulic resistance coefficient. The physical meaning of this coefficient is that it indicates the hydraulic resistance per unit line cross section area. The use of this coefficient is convenient for analysis of a complex circuit.

In hydraulic calculations we often encounter the dynamic pressure q

$$q=\frac{\gamma V^2}{2g}, \tag{27}$$

which is also conveniently represented in the form

$$q=c^2\frac{1}{d^4}\gamma W^2.$$

When necessary, the sum $\Delta p_h + q$ can be calculated from the formula

$$\Delta p_r + q = c^2 \frac{\zeta_0}{d^4} \gamma W^2 + c^2 \frac{1}{d^4} \gamma W^2 = c^2 \frac{\zeta_0 + 1}{d^4} \gamma W^2 = c^2 k_c^2 \gamma W^2,$$

where

$$k_c^2 = \frac{\zeta_0 + 1}{d^4} = k^2 + \frac{1}{d^4}.$$

Complex Circuit

Let us examine a complex circuit consisting of lines of different diameter connected in series (Figure 27). The flowrate through each line is the same

$$W_1 = W_2 = W_i = W.$$

To determine the circuit hydraulic resistance graphically (Figure 28), we use (24) to find the resistances of lines 1 and 2 (curves Δp_1 and Δp_2). For several values of the flowrates we sum the hydraulic resistances of the lines by adding the ordinates for the same abscissa. The resulting curve shows the overall value of the hydraulic resistance $\Delta p_{\Sigma_{ser}}$.

Thus

$$\Delta p_{\Sigma_{\text{посл}}} = \Delta p_1 + \Delta p_2 + \ldots + \Delta p_i$$

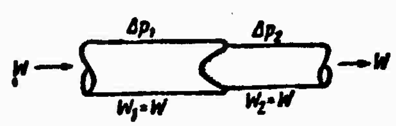


Figure 27. Series connection of lines

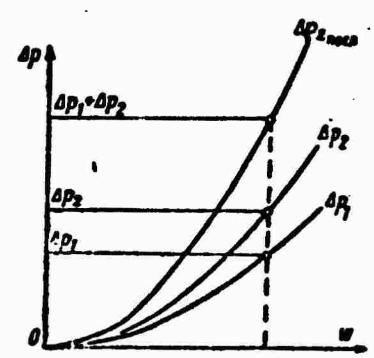


Figure 28. Graphical determination of hydraulic resistance for series connection of lines

or

$$c^2 k^2_{\Sigma_{\text{посл}}} \gamma W^2 = c^2 k_1^2 \gamma W^2 + c^2 k_2^2 \gamma W^2 + \ldots + c^2 k_i^2 \gamma W^2.$$

After cancellation, we obtain

$$k^2_{\Sigma_{\text{посл}}} = k_1^2 + k_2^2 + \ldots + k_i^2,$$

$$k_{\Sigma_{\text{посл}}} = \sqrt{k_1^2 + k_2^2 + \ldots + k_i^2}. \qquad (28)$$

For <u>parallel</u> connection of lines (Figure 29), the flowrate at the branch point equals the sum of the flowrates

$$W = W_1 + W_2 + \ldots + W_i.$$

To determine the circuit hydraulic resistance we plot on a graph (Figure 30) the hydraulic resistance curves for each line (curves Δp_1 and Δp_2). To combine the flowrates we take the flowrates for several values of the hydraulic resistances and sum them by adding the abscissas for the same ordinate. The curve $\Delta p_{\Sigma_{\text{par}}}$ indicates the value of the hydraulic resistance for parallel connection of the lines. When adding the flowrates, we observed the condition that the hydraulic resistances be equal

$$\Delta p_{\Sigma_{\text{пар}}} = \Delta p_1 = \Delta p_2 = \Delta p_i = \Delta p.$$

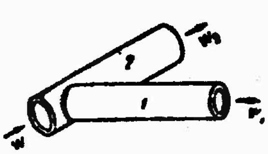


Figure 29. Parallel connection of lines

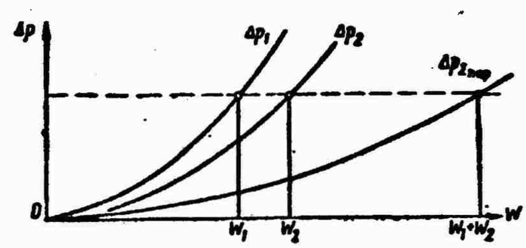


Figure 30. Graphical determination of hydraulic resistance for parallel connection of lines

We obtain the flowrate from (24)

$$W=\sqrt{\frac{\Delta p}{c^2k^2\gamma}}=\frac{1}{ck}\sqrt{\frac{\Delta p}{\gamma}}.$$

The condition for combining flowrates in the parallel connection case permits writing

$$\frac{1}{ck_{\Sigma_{\text{пар}}}}\sqrt{\frac{\Delta p}{\gamma}}=\frac{1}{ck_1}\sqrt{\frac{\Delta p}{\gamma}}+\frac{1}{ck_2}\sqrt{\frac{\Delta p}{\gamma}}+\ldots+\frac{1}{ck_i}\sqrt{\frac{\Delta p}{\gamma}}.$$

After cancellation we obtain

$$\frac{1}{k_{\Sigma_{\text{пар}}}}=\frac{1}{k_1}+\frac{1}{k_2}+\ldots+\frac{1}{k_i}$$

and finally

$$k_{\Sigma_{\text{пар}}}=\frac{1}{\frac{1}{k_1}+\frac{1}{k_2}+\ldots+\frac{1}{k_i}}. \qquad (29)$$

We denote $\rho = \frac{1}{k}$. The quantity ρ can be called the hydraulic conductance. Then

$$\rho_{\Sigma_{\text{пар}}}=\rho_1+\rho_2+\ldots+\rho_i. \qquad (30)$$

We see from the condition of equality of the hydraulic resistances that

$$c^2k^2_{\Sigma_{\text{пар}}}\gamma W^2=c^2k_1^2\gamma W_1^2=c^2k_2^2\gamma W_2^2=c^2k_i^2\gamma W_i^2.$$

After transformations we obtain

$$k_{\Sigma_{\text{пар}}}W=k_1W_1=k_2W_2=k_iW_i.$$

Consequently

$$kW=\text{const} \qquad (31)$$

or

$$\frac{W}{\rho}=\text{const.}$$

For the overall flowrate $W = W_1 + W_2$ in the case of parallel connection of two lines and the reduced hydraulic resistance coefficients k_1 and k_2 (hydraulic conductances ρ_1 and ρ_2) of the branches

$$k_1W_1=k_2W_2=\text{const} \quad \text{or} \quad \frac{W_1}{\rho_1}=\frac{W_2}{\rho_2}=\text{const.}$$

Hence

$$\frac{W_1}{W_2}=\frac{\rho_1}{\rho_2}=\frac{k_2}{k_1}. \tag{32}$$

Writing the derivative ratios, we obtain

$$\frac{k_1+k_2}{k_2}=\frac{\rho_1+\rho_2}{\rho_1}=\frac{W_1+W_2}{W_1}=\frac{W}{W_1};$$

and

$$\frac{k_1+k_2}{k_1}=\frac{\rho_1+\rho_2}{\rho_2}=\frac{W_1+W_2}{W_2}=\frac{W}{W_2}.$$

Then we can obtain the <u>flowrate division between the branches of the parallel junction</u> in the form

$$W_1=\frac{k_2}{k_1+k_2}W=\frac{\rho_1}{\rho_1+\rho_2}W; \tag{33}$$

$$W_2=\frac{k_1}{k_1+k_2}W=\frac{\rho_2}{\rho_1+\rho_2}W. \tag{34}$$

Since we know how to find the reduced hydraulic resistance coefficients for series and parallel junctions, there is no difficulty in finding them for a circuit consisting of a <u>mixed connection</u> of lines (Figure 31). We draw the contour k_{bc}, enclosing the elements k_b and k_c, and the contour k_{circ}, which encloses the contour k_{bc} and the element k_a (the designations of the contour and elements correspond to the values of the reduced hydraulic resistance coefficients).

The element k_a and the contour k_{bc} are connected in series, so that

$$k_{\text{маг}}=\sqrt{k_{\text{а}}^2+k_{\text{б.в}}^2}.$$

We can write for the contour k_{bc}, as for a parallel connection, the expression

$$k_{\text{б.в}}=\frac{1}{\frac{1}{k_{\text{б}}}+\frac{1}{k_{\text{в}}}}.$$

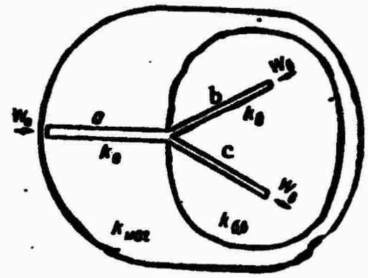


Figure 31. Mixed connection of lines

The reduced hydraulic resistance coefficient of the circuit for the given mixed line junction is

$$k_{\text{маг}}=\sqrt{k_{\text{а}}^2+\left(\frac{1}{\frac{1}{k_{\text{б}}}+\frac{1}{k_{\text{в}}}}\right)^2}. \qquad (35)$$

Inertial Pressure Losses

Inertial pressure losses Δp_i arise in circuits when the flight vehicle moves with the acceleration j. These losses for a line segment can be expressed by the ratio of the inertial force mj to the line internal section area f

$$\Delta p_{\text{и}}=\frac{mj}{f}=\frac{Gj}{gf}=\frac{Gn}{f}\ \text{N/m}^2 \qquad (36)$$

where $m = \frac{G}{g}$ — is mass, N · sec^2/m;

G — is weight, N;

j — is acceleration, m/sec^2;

$n = j/g$ — is the load factor.

Replacing the fluid weight by the specific weight γ and volume E and representing the volume in terms of the line length l and internal cross section area f, we obtain

$$\Delta p_{u} = \frac{Gn}{f} = \frac{\gamma nE}{f} = \gamma nl.$$

Under the influence of the load factors n_x, n_y, n_z the inertial pressure losses will be

$$\Delta p_{u_x} = \gamma n_x \sum' l_x; \quad (37)$$
$$\Delta p_{u_y} = \gamma (n_y - 1) \sum' l_y; \quad (38)$$
$$\Delta p_{u_z} = \gamma n_z \sum' l_z. \quad (39)$$

where $\Sigma' l_x, \Sigma' l_y, \Sigma' l_z$ — are the total projections on the x, y, z axes of all the line lengths, obtained with account for the fluid direction of motion. Thus, for the case shown in Figure 32, $\Sigma' l_y = y_2 - y_1$.

The subtraction of one from n_y in (38) is explained by the fact that n_y = 1 in steady level flight and there are no inertial pressure losses.

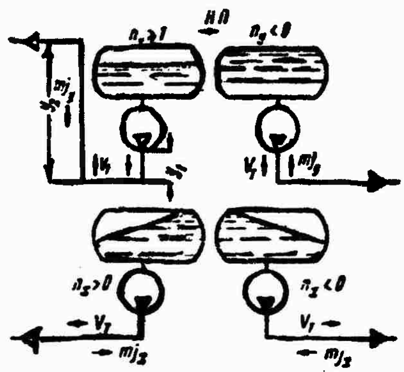


Figure 32. Cases in which inertial pressure losses are considered in selection of boost pump

The inertial pressure losses are taken into account if the fluid tends to separate from the component (pump) which it is entering, i.e., when the inertial forces are directed opposite the direction of motion of the fluid at the inlet to the component in question. In these cases the magnitudes of the inertial pressure losses are always assigned a positive value, regardless of the sign obtained in the calculation.

Cavitational Phenomena

Cavitation is the process of vapor and gas bubble formation in a low-pressure zone and subsequent contraction of these bubbles in a high-pressure zone.

The physical meaning of cavitation is that, when a liquid flows through a pipe or flows past pump elements, a local pressure reduction takes place to a pressure at which there is intense release of the gases and vapors dissolved in the liquid (local cold ebullition). When the pressure is increased, the vapors and gases begin to dissolve in the liquid (condensation). The liquid rushes into the free space which is formed. The condensation process is accompanied by hydraulic impact. The number and force of these impacts are quite large for a single-component liquid (water, alcohol).

As a result of the impact loading, there is a reduction of the efficiency of hydraulic machines and onset of cavitational erosion. Cases have been recorded in which the operation of marine propellers and hydroturbines in the cavitational regimes led to their failure.

The nature of cavitational phenomena development for the multicomponent liquids consisting of light and heavy fractions (gasoline, kerosene) differs somewhat from cavitation in the single-component liquid. The light fractions have higher values of the saturated vapor pressure than do the heavy fractions; therefore the light fractions boil first in the case of local pressure reduction, and then the heavy fractions boil. Upon increase of the pressure the vapors of the heavy fractions condense first and then the vapors of the light fractions. As a result of this the cavitation process proceeds gradually, without impacts, in the multicomponent liquids.

Cavitation can occur in aircraft powerplant systems in connection with the external pressure reduction as the flight altitude is increased. In the initial stage the vapor phase is represented by fine bubbles (Figure 33a). Then the bubbles increase in size and in a horizontal line they will travel in the upper part of the cross section (Figure 33b). It is possible that the vapor and liquid phases will separate (Figure 33c) and the stream will split.

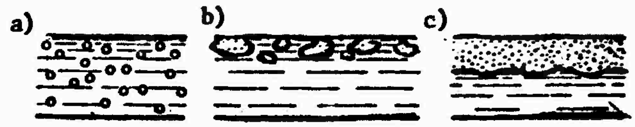


Figure 33. Formation of two-phase flow (liquid and vapor)

The highest pressure of the vapors located above a liquid, established during vapor release in a closed vessel at a given temperature, is called the saturated vapor pressure and is denoted by p_t. For the single-component liquid this pressure depends only on the temperature and physical properties of the given liquid and is independent of the volumetric ratio of the vapor and liquid phases. The saturated vapor pressure of the multicomponent liquid depends not only on the temperature but also on the ratio of these phases. The saturated vapor pressure increases with reduction of the volume fraction occupied by the vapor phase. In laboratory tests of aircraft fuels a standard ratio of the vapor and liquid phases, equal to 4/1, has been adopted and is denoted by $P_{t4/1}$.

With increase of the liquid temperature, the saturated vapor pressure of the single and multicomponent liquids increases; however the increase is different for different liquids. In order to be able to characterize the saturated vapor pressure of a liquid by a single number the temperature 37.8° C = 100° F has been arbitrarily adopted. The pressure at this temperature is termed the Reid vapor pressure.

Cavitation phenomena in the fuel lines of aircraft powerplants present more hazard with regard to stoppage of fuel flow to the engines than danger of line failure. This is associated with the solubility of air in the fuel. In the petroleum fuels there is some amount of air (10 - 20% by volume), which is directly proportional to the ambient pressure and inversely proportional to the fuel specific weight, surface tension, and viscosity. Moreover, there are also dissolved gases (propane, butane, and so on).

As the flight altitude increases, the atmospheric pressure decreases and the pressure in the fuel system tanks and lines decreases. A large quantity of air and gaseous inclusions are released into the space above the fuel and carry with them fuel vapors. If the external pressure is greater than the saturated vapor pressure of the fuel, vaporization of fuel from the surface does not have any significant effect on the size and rate of release of the air bubbles. When the external pressure is lower than the saturated vapor pressure of the fuel, internal vaporization — boiling of the fuel — begins. The higher the saturated vapor pressure of the fuel, the earlier boiling will begin.

The cavitation phenomena which occur in the liquid lead to separation of the flow in lines into liquid and gaseous streams. As a result the flow through the line is mixed, which leads to pressure pulsations, oscillations, and interruptions in the fuel supply which may result in involuntary shutdown of the engine. Moreover, in the presence of cavitation the hydraulic resistance increases and heat transfer decreases.

Cavitation phenomena are most frequently observed at the inlet to pumps. The lower the inlet pressure, the lower will be the pressure differential created by the pump and its output. The cavitation characteristics of a pump are determined from the cavitation curves.

The cavitation curves, obtained experimentally, establish the relationship between the pump output (Figure 34) and differential pressure (Figure 35) and the pump inlet pressure. These curves are presented for a given liquid for several values of pump shaft speed and temperature. If the pump output is being determined, a constant differential pressure is maintained in the tests; to find the differential pressure developed by the pump, a constant output is maintained.

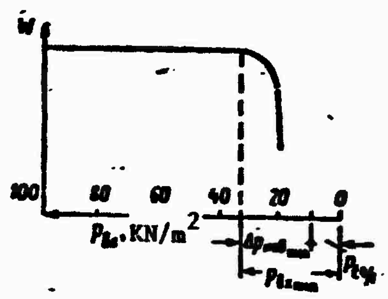


Figure 34. Pump cavitation curve for constant values of shaft speed, differential pressure, and liquid temperature

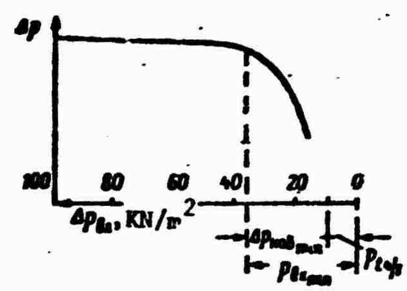


Figure 35. Pump cavitation curve for constant values of shaft speed, output, and liquid temperature

The inlet pressure is plotted along the abscissa axis of the cavitation curves, so that increasing inlet pressures are plotted toward the coordinate origin. Thus, the magnitude of the underpressure is plotted in the opposite direction (away from the origin). This technique for laying out the abscissa axis scale means that the nature of the curves will remain the same if the abscissa scale shows the flight altitude as well as the pump inlet pressure. Therefore the pump cavitation curve reflects its altitude performance.

With low inlet pump pressure the cavitation is so extensive that cutoff of pump operation takes place — decay of the output and differential pressure to zero. Theoretically this should occur at an ambient pressure equal to the saturated vapor pressure. However, in practice pump cutoff occurs at an inlet pressure greater than the saturated vapor pressure. This is explained by the fact that the actual pressure of the liquid in the pump entrance chamber is less than the measured inlet pressure by the magnitude of the hydraulic resistance of the pump suction section. Moreover, in fuel pumps the actual ratio of the fuel vapor and liquid phases does not correspond to the standard 4/1 ratio for which the pump tests are conducted.

The drooping branches of the pump cavitation curves cannot be defined experimentally with sufficient accuracy, since the pump operating regimes in this region are unstable and may become self-oscillatory. This is associated with the fact that random decrease of the pump inlet pressure leads to reduction of the output, as a result of which there is reduction of the velocity and the pump inlet pressure increases. The latter increases the output and then the pump inlet pressure decreases, which again leads to reduction of the output.

In order to obtain the stable required differential pressures created by the pump and an output at which normal liquid delivery is assured, it is necessary to create at the pump inlet an excess pressure which exceeds the saturated vapor pressure of the liquid. This excess pressure is usually called the required cavitation margin Δp_{cav}. Consequently the pump noncavitation operating condition is

$$p_{вх} \geqslant p_{t\,4/1} + \Delta p_{кав}.$$

The minimal required pump inlet pressure is

$$p_{вх\,min} = p_{t\,4:1} + \Delta p_{кав\,min}. \qquad (40)$$

The minimal required cavitation pressure margin depends on the pump construction and location, amount of pump wear, pump shaft speed, flight vehicle rate of climb, and air content in the fuel. All these factors cannot be calculated exactly, and therefore the magnitude of the required minimal cavitation pressure margin is usually indicated by the manufacturer on the basis of tests. If this value is not specified by the manufacturer, it can be established in several ways.

1. The magnitude of the minimal required cavitation pressure margin is taken from the cavitation curve, depending on the required values of the output or differential pressure created by the pump, provided these values fall on the stable rather than the cutoff segment of the cavitation curve.

2. The magnitude of the required cavitation pressure margin for centrifugal pumps is found using approximate formulas, on the basis that it can be expressed as a fraction of the differential pressure Δp_p created by the pump

$$\Delta p_{\text{кав.min}} = \sigma \Delta p_{\text{н}},$$

where σ — is the cavitation index.

The cavitation index for centrifugal pumps is a function of the rotor specific speed n_s

$$\sigma = A n_s^{4/3}. \tag{41}$$

By <u>specific speed</u> we mean the rotor speed of a reference pump which is geometrically similar to the given pump and has dimensions such that for a head of 1 m H_2O the output W is 0.075 m^3/sec.

In hydraulics courses the expression for the specific speed is given in the form

$$n_s = 3{,}65 \frac{60 n \sqrt{W}}{\left(\frac{\Delta p_{\text{н}}}{\gamma}\right)^{3/4}}, \tag{42}$$

where n — is the pump rotor speed, rps.

Substituting the value of the specific speed into (41), we obtain

$$\sigma = An_s^{4/3} = A\left[\frac{3,65\cdot 60n\sqrt{W}}{\left(\frac{\Delta p_н}{\gamma}\right)^{3/4}}\right]^{4/3} = \frac{3,65^{4/3}A(60n)^{4/3}W^{2/3}\gamma}{\Delta p_н} =$$

$$= \frac{10(60n)^{4/3}W^{2/3}\gamma}{C^{4/3}\Delta p_н},$$

where

$$\frac{10}{C^{4/3}} = 3,65^{4/3}A.$$

Then the required cavitation pressure margin of the centrifugal pump can be expressed in the form

$$\Delta p_{кав_{min}} = \sigma\,\Delta p_н = \frac{10(60n)^{4/3}W^{2/3}\gamma}{C^{4/3}}\ н/м^2. \qquad (43)$$

The value of the cavitation specific speed suggested by Professor Rudnev [49] is

$$C = \frac{10^{3/4}\cdot 60\,n\sqrt{W}\,\gamma^{3/4}}{(\Delta p_{кав_{min}})^{3/4}} = \frac{5,62\cdot 60n\sqrt{W}}{\left(\frac{\Delta p_{кав_{min}}}{\gamma}\right)^{3/4}} = 1,54\,\frac{n_s}{\sigma^{3/4}}.$$

For centrifugal pumps with an axial prestage (ahead of the centrifugal impeller), the cavitation specific speed varies from 700 to 2200 [8, 34]. To ensure reliable operation of aircraft powerplant systems, we can take values of the coefficient C in the range 1000 - 1400.

Reduction of the required cavitation pressure margin is achieved by reducing the pump rotor speed and pump output. The use of duplicate pumps in some lines, in addition to reliability of the supply in case of failure of one of the pumps, reduces the required cavitation pressure margin for two pumps operating in parallel.

3. The statistical data of Table 3 are used. We see from this table that with increase of the differential pressure created

by the pump the absolute magnitude of the minimal required cavitation pressure margin increases, but the value of the cavitation index decreases.

TABLE 3. MINIMAL REQUIRED PUMP CAVITATION PRESSURE MARGIN

Pump type	Pump location	Differential pressure KN/m^2	Minimal required cavitation pressure margin KN/m^2	Cavitation index σ
Boost and transfer	Wing	100 - 150	10 - 25	0.15 - 0.25
Boost	Engine	500 - 600	30 - 80	0.05 - 0.15
Fuel supply to injection nozzles	Engine	6000 - 8000	150 - 300	0.02 - 0.05

Centrifugal Pump Characteristics

In addition to the cavitation curves (see Figures 34 and 35), one of the basic characteristics is the head curve, which establishes the relationship between the differential pressure created by the pump and the output for constant values of pump shaft speed and liquid temperature (Figure 36).

For centrifugal pumps with electric motor drive, we can define the power required

$$N_{потр} = IU \text{ W},$$

where I — is the current consumed, A;
U — is the voltage, V.

The pump useful power is defined by the formula

$$N_{пол} = \Delta p_{н} W \ \text{W},$$

where Δp_p — is expressed in N/m^2.

The pump operating regime is defined by the point of intersection of the head curve with the line hydraulic characteristic. Different operating regimes can be obtained for a constant pump head curve by varying the line hydraulic characteristic (throttling), and for a constant line hydraulic characteristic by varying the pump shaft speed (Figure 37).

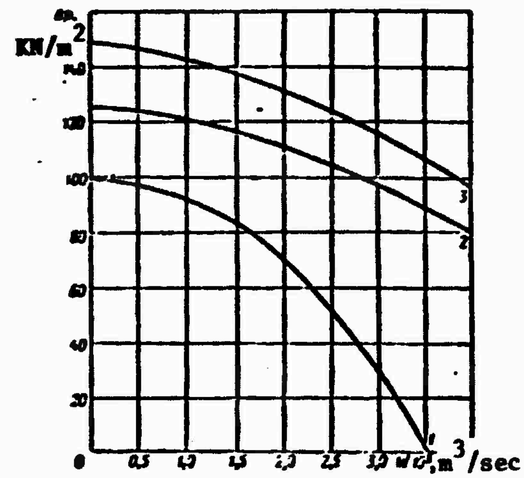


Figure 36. Head curves of centrifugal boost pumps:
1 — EtsN-45; 2, 3 — ETsN-T and ETsN-10 (at rated conditions)

Centrifugal pump output and differential pressure depend on the rotor diameter d and shaft speed n. From the similarity formulas for centrifugal pumps, we have

$$\frac{W}{W_1} = \frac{nd^3}{n_1 d_1^3};$$

$$\frac{\Delta p}{\Delta p_1} = \frac{n^2 d^2}{n_1^2 d_1^2}.$$

Change of the rotor diameter leads to new values of the pump weight and size. However, change of the shaft speed, within permissible limits, makes it possible in many cases to use an available pump. This property of the centrifugal pump is used for its regulation.

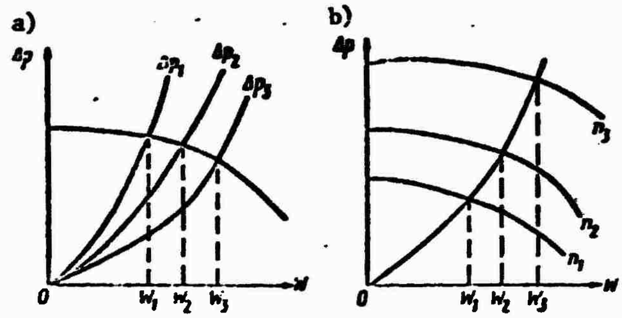


Figure 37. Centrifugal pump operating regime variation: a — variation of line hydraulic characteristic; b — variation of pump shaft speed

For $d = d_1$

$$\frac{W}{W_1}=\frac{n}{n_1};\quad \frac{\Delta p}{\Delta p_1}=\frac{n^2}{n_1^2}=\frac{W^2}{W_1^2}.$$

Hence

$$\Delta p=\frac{\Delta p_1}{W_1^2}W^2. \tag{44}$$

If the values of Δp_1 and W_1 are known, by specifying the output W we can find the differential pressure Δp and plot from these points the curve of similar regimes (corresponding to the condition of constant hydraulic and volumetric efficiencies of the pumps) in Figure 38, which shows the initial pump head curve.

The perpendicular dropped to the abscissa axis from the point of intersection of the line of similar regimes with the head curve 1 yields the value W_0 for the shaft speed n_0. This makes it possible to find the shaft speed n_1 which the pump must have to provide the output W_1 and the differential pressure Δp_1.

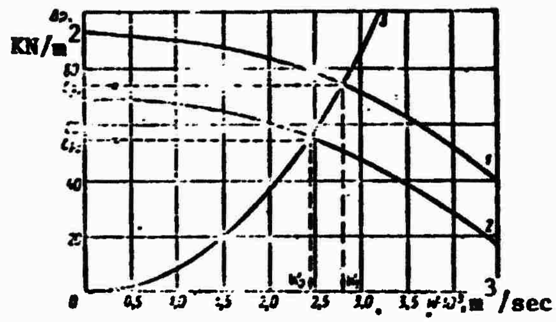


Figure 38. Head curves of centrifugal boost pump for different rotor speeds; 1 — n_0 = 4800 rpm; 2 — n_1 = 5560 rpm; 3 — curve of similar regimes

Since

$$\frac{n_1}{n_0}=\frac{W_1}{W_0}, \text{ then}$$

$$n_1=\frac{W_1}{W_0}n_0. \tag{45}$$

The pump head curve for the shaft speed n_1 can be obtained by scaling each point of the initial head curve using the formulas

$$W_{1i}=\frac{n_1}{n_0}W_{0i}; \tag{46}$$

$$\Delta p_{1i}=\left(\frac{n_1}{n_0}\right)^2\Delta p_{0i}. \tag{47}$$

For the case of parallel operation of several pumps the overall head curve is obtained just as for the case of parallel connection of hydraulic lines, i.e., we combine the conductances for a constant pressure (Figure 39).

Figure 39. Head characteristic of pumps connected in parallel

CHAPTER III

FUEL SYSTEMS

Fuel systems are designed to provide the required amount of fuel to the engines. They are subject to the following requirements:

1. Reliable fuel supply to engines in all flight regimes.

2. Fire safety.

3. Tank capacity adequate for the required amount of fuel.

4. Automatic and complete fuel feed from tanks in given sequence while maintaining the airplane center of gravity (c.g.) in an acceptable range.

5. Pressure refueling for flight vehicles with total tank capacity more than 6000 liters.

6. Fuel dumping in flight for vehicles having landing weight or c.g. limitations.

7. Complete ground defueling capability.

8. Reliable and convenient monitoring of fuel system operation on the ground and in flight.

P'ston engines use B-70, B-91/115, and B-95/130 (GOST 1012-54) gasolines. Some engines use SB-78 gasoline containing 20 - 25% B-91/115 and 80 - 75% B-70 (TU4-60). Turbojet and turboprop engines operate on kerosene grades T-1, TS-1 (GOST 10,227-62), T-5 (GOST 9145-59), T-6 and T-7 (GOST 12,308-66).

In analyzing the operation of fuel systems, we encounter the following fuel characteristics: density (specific weight), saturated vapor pressure, viscosity, specific heat, and thermal conductivity (see Appendices 3 - 7).

The fuel system is a complex of subsystems: engine fuel feed, fuel tank venting, automatic control of fuel flowrate, and fuel measurement. The engine fuel feed system consists of circuits which provide for fuel supply to the engines, refueling, and drainage.

Engine Fuel Supply Circuits

Fuel Supply Schemes

The choice of a rational scheme for supplying fuel to the engines is affected by the flight vehicle mission and configuration, flight regimes, type and number of engines, grade of fuel used, measures to ensure safety, flight ceiling, and flight vehicle reliability. The problem in developing a rational scheme for feeding fuel to the engines lies in the need for locating a large amount of fuel in a limited space, maintaining an established range of c.g. variation, requirements for ensuring trouble-free operation of the engines over a wide range of flight speeds and altitudes, and the introduction of automatic equipment to provide a specified fuel feed sequence and monitoring fuel system operation.

We shall examine some schemes used to supply fuel to the engines (Figure 40). Fuel feed from the tank 5 by means of the pump 6 makes it possible to develop a sufficiently high pressure at the inlet to the engine driven pump to ensure the required altitude performance. The tank airspace is connected with the ambient air by means of the vent intake 1. This system for feeding fuel from the

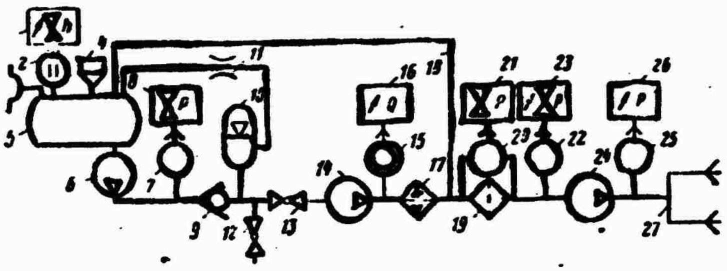


Figure 40. Fuel feed to engines

tanks is termed an open system. It is the basic system for use on civil aircraft. The tank is filled with fuel through the filler neck or fitting 4. The fuel level meter with transmitter 2 and indicators 3 is used to determine the amount of fuel in the tank and provide emergency low-level indication.

The pumps which are used to feed fuel from the tanks and create pressure at the inlet to the engine pump are termed the airframe boost pumps (ABP). The purpose of the ABP is to overcome all the resistances along the path from the tank to the engine pump and provide the required pressure at the inlet to the engine pump. Multistage boosting is used on the latest aircraft to provide reliable fuel feed to the engines. Usually a single airframe boost pump 6 and a single engine boost pump (EBP) 14 are used. In this case the ABP provides the required pressure at the inlet to the EBP, and the latter provides the required pressure at the inlet to the main engine pump (MEP) 24.

The advantage in feeding fuel from the tanks by means of ABP is that the tanks are not pressurized and their weight can be kept low. Fuel can be pumped from a tank located below the engine. In many cases the ABP operating regime is regulated to maintain the required pressure. A definite sequence of ABP activation and deactivation permits a programmed sequence of fuel feed from the tanks. The power required to drive the ABP is one of the drawbacks in their use. Increase of the thrust (power) of the gas turbine engines and the corresponding increase of the fuel flowrate increase the power, weight, and size of the ABP.

The pressure developed by the ABP must be greater than the minimal acceptable value for which the pressure warning transmitter 7 is set. When the pressure is adequate, either the red light 8 goes out or the green light 8 comes on at the pilot's instrument panel, depending on the configuration of the system for indicating the pressure downstream of the ABP.

The check valve 9 ensures the required direction of fuel flow. For example, if there are two ABP in a single tank and one of them fails, the fuel picked up by the other would return through the inoperative pump into the tank. However, the check valve located downstream of the inoperative ABP blocks fuel flow into the tank. Check valves operate similarly when the fuel accumulator 10 becomes operative, when the crossfeed valve 12 is opened, and in other cases. Air from the accumulator fuel chamber flows through the restrictor 11 into the tank.

The fire cock 13 shuts off the fuel flow. The flowmeter transmitter 15 provides an indication of the fuel flowrate on the indicator 16. The fuel/oil radiator 17 is installed in the circuit to cool the oil using fuel as the coolant and at the same time provides heating for the fuel. This improves atomization and protects the filter 19 against freezing. If less fuel flow is required to the engine than for cooling the oil in the fuel/oil radiator, part of the fuel will bypass the radiator and return to the tank through the bypass line 18. The pressure warning transmitter 20 together with the indicator 21 provides indication of clogging (freezing) of the filter 19. The pressure transmitter (or pressure switch) 22 sends a signal to the instrument 23, which indicates the value (or presence) of the pressure ahead of the MEP. The pressure transmitter 25 downstream of the MEP and the pressure gage 26 make it possible to determine the magnitude of the pressure at the inlet to the nozzle manifold 27.

Gravity fuel feed from the tanks (Figure 41) is possible but relatively ineffective. This feed technique is used on aircraft with comparatively low-power PE, where the required pressure at the engine pump inlet is not high. On aircraft with engines developing high thrust (power) gravity fuel feed from the tanks is used for gravity transfer of fuel from a higher tank to a lower tank. This is possible when the wing has positive dihedral.

Fuel feed from the tanks by expulsion can be accomplished by compressed air or inert gases (Figure 42). The tank air space is

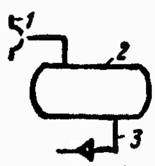


Figure 41. Fuel feed from aircraft tank:

1 — air intake from atmosphere; 2 — tank; 3 — fuel line to engine pump

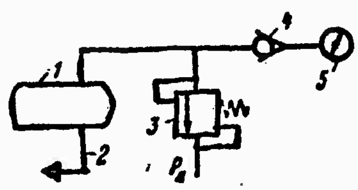


Figure 42. Fuel feed from tank by expulsion:

1 — tank; 2 — fuel line to engine pump; 3, 4 — relief and check valves; 5 — air intake from engine

isolated from the ambient air (closed feed takes place). The advantages of closed feed are: high flight altitude, absence of fuel pumps on the airframe (and sometimes on the engine also), possibility of pressure regulation, absence of venting system, no losses owing to fuel evaporation, and no energy expended to drive pumps. However there are significant disadvantages: heavy pressurized tanks and low survivability of the tanks when damaged. Fuel feed from the tanks by expulsion is not used on today's civil aircraft; however in certain cases pressurization of the fuel tank using a slight excess pressure (15 - 30 KN/m^2) may be used. This excess pressure is obtained from the engine compressor (through a restrictor) or from ram pressure.

Large tanks are required if the amount of fuel carried is large. Difficulties in installing such tanks make it necessary to use small tanks but their number increases correspondingly. In order to organize efficient fuel feed to the engines with small hydraulic resistances, low line weight, and to ensure the required range of c.g. travel, the tanks are combined into groups, usually by connecting them in series (Figure 43). Thus a group of tanks can be considered as a single tank with baffles.

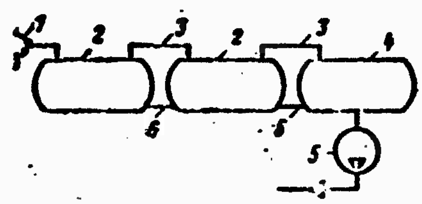


Figure 43. Connecting tanks into a group:

1 — air intake from atmosphere; 2 — tanks; 3 — vent line; 4 — service tank; 5 — boost pump; 6 — fuel line

The tank on which the boost pump is installed to supply fuel to the engines is called the service tank, and the line running from the service tank to the engine is called the engine intake line. The tank groups can, in turn, be connected in parallel or series.

The fuel supply system with groups of tanks connected in parallel (Figure 44) makes it possible to control fuel usage in accordance with any program. The pump of one tank (one group) can operate in a regime which differs from the pump operating regime of another tank (another group). In this case the check valve located at the pump with the lower operating regime closes and ensures fuel feed only from the tank with the pump operating in the higher regime. The survivability of the fuel system is better with this grouping of tanks. This scheme involves the installation of a boost pump in each group of tanks, which increases the structural weight and requires a complex feed system.

The fuel supply system with groups of tanks connected in series (Figure 45) is constructed so that the fuel feeds from each group into a service tank, from which the fuel is supplied to the engine. This scheme permits reduction of the system weight, since fewer pumps are required, the number of lines is reduced, and a simple control system can be used. However, the presence of the service tank reduces the survivability of the fuel system.

In the fuel systems which feed the fuel from groups of tanks connected in series, the fuel is transferred by gravity feed or by a pump into the service tank. Gravity transfer or pressure transfer lines are provided for these operations. The transfer pumps (TP)

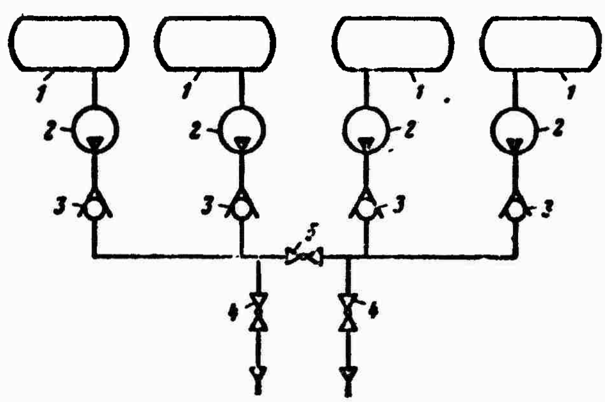


Figure 44. Fuel feed to two engines with tanks (tank groups) connected in parallel:

1 — tanks; 2 — boost pumps; 3 — check valves; 4,5 — fire shutoff and crossfeed valves

direct the fuel from one tank to another with only the pressure necessary to overcome the resistance in the transfer segment. Therefore these pumps produce less pressure differential than the ABP. This reduces the weight and power required. A float-type level-limiting valve is installed in the service tank to prevent overfilling.

Independent, centralized, and combined systems are used to supply fuel to several engines. The selection of the particular fuel supply system depends on the aircraft layout and the number of engines.

The <u>independent systems</u> (see Figures 44 and 45) supply fuel to each engine of the aircraft from definite groups of tanks.

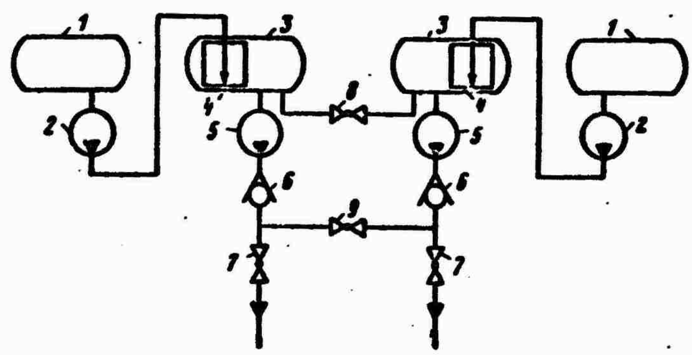


Figure 45. Fuel feed to two engines with tanks (tank groups) connected in series:

1 — tanks; 2, 5 — transfer and boost pumps; 3 — service tanks; 4, 6 — safety float and check valves; 7, 8, 9 — fire shutoff, tank interconnect, and crossfeed valves

The <u>centralized system</u> (Figure 46) is arranged so that fuel from a single service tank (or from several tanks through a selector valve) is supplied to all engines.

The <u>combined systems</u> are a combination of the independent and centralized systems. Figure 47 shows a combined system consisting

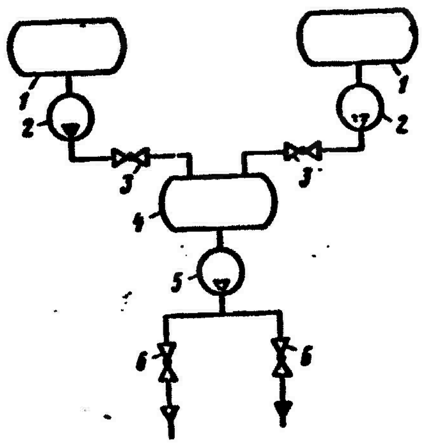


Figure 46. Centralized fuel feed to two engines:

1 — tanks; 2, 5 — transfer and boost pumps; 3, 6 — tank shutoff and fire shutoff valves; 4 — service tank

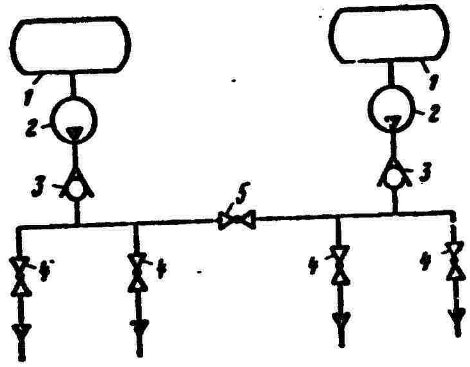


Figure 47. Combined fuel feed to four engines:

1 — service tanks; 2 — boost pumps; 3 — check valves; 4, 5 — fire shutoff and crossfeed valves

of two independent systems (left and right), but each of these systems provides centralized fuel flow to both engines.

On aircraft with an odd number of engines the fuel flow to the central engine, usually located in the aft fuselage, is provided by the ABP of two independent systems. These pumps may differ in their head characteristics within their tolerance limits, which can lead to different fuel flow to the center engine from the pumps of the independent systems with tanks connected in series, and this then leads to different fuel levels in the service tanks. A similar situation is observed when opening the crossfeed valve and supplying fuel from the pumps of the two independent systems to a single engine. Even if the pump characteristics are the same, the fuel flowrates from the two pumps will be different because of the different hydraulic resistances of the engine intake lines.

Equalization of the fuel level in the service tanks of the independent systems can be accomplished by gravity transfer if the tanks are interconnected by a gravity crossfeed line (see Figure 45). If the tank interconnect valve 8 is open, gravity crossfeed will take place when the fuel level in one tank reaches a definite height above the fuel level in the other tank.

The following measures are taken to improve the reliability of the engine fuel feed system.

Fuel crossfeed to the engines is used on aircraft with several tanks (groups of tanks) and having two or more engines. The engine intake lines are interconnected by a crossfeed line downstream of the ABP (see Figures 44, 45, and 47). In case of failure of one of the engines, with the crossfeed valve open fuel will feed to the operating engine not only from its own intake line but also from the intake line of the inoperative engine. If the ABP of one of the independent intake lines fails, operation of the engine receiving fuel through this line will be provided by means of the ABP in the other intake line if the crossfeed valve is open. Therefore the possible ABP output exceeds the maximal fuel flowrate of a single engine. However, fuel will not feed from the tanks connected with the intake line of the failed ABP, and this affects the aircraft c.g. location, flight range and duration. The crossfeed line can also be used to equalize the fuel quantity remaining in the service tanks of the independent engine intake lines.

Duplication of the ABP involves the installation of two pumps operating in parallel, each of which has an output sufficient for independent supply of fuel to the engines. When both pumps are operating, each supplies about half the fuel flow to the engines, which reduces the required cavitation pressure margin and increases the ceiling. However, duplication increases the structural weight and requires additional power to drive the pumps.

Airframe boost pump <u>reservation</u> involves provision for activation of a second pump in case one pump fails. The second pump may have a different type of drive in order to improve the survivability of the fuel system.

<u>Pickup wells</u> with check valves are used in the tanks to prevent uncovering the ABP during airplane maneuvers and under the action of accelerations. These wells create a reserve fuel volume as the fuel flows away from the boost pump intake (Figure 48). Boost and transfer pumps can also be installed in the pickup wells.

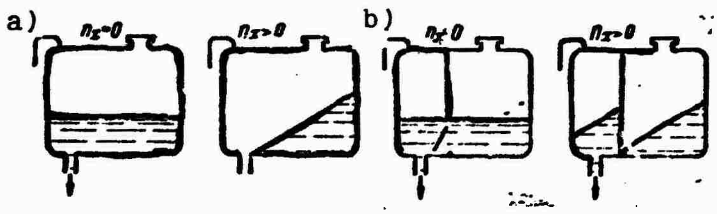


Figure 48. Fuel tanks:
a — without baffle; b — with baffle and check valve

When the pressure in certain segments of the intake line increases above a definite value, the <u>pressure relief lines</u> bypass fuel from the segment with the high pressure into a low-pressure segment, for example into the tank (Figure 49). The pressure may increase after shutting the engine down with the fire shutoff cock. In this case the flow of cold fuel terminates, and as the fuel is heated in the line segments which pass near the still hot engine sections, it expands and causes deformation of the line

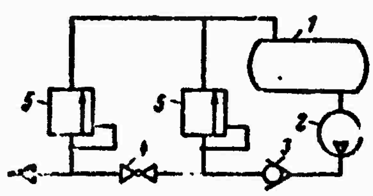


Figure 49. Relief lines for high-pressure fuel bypass:
1 — tank; 2 — boost pump; 3, 5 — check and relief valves; 4 — fire shutoff valve

elements. Moreover, cocks with lever control of the shutoff sleeve do not operate satisfactorily against high backpressure. Relief of the high pressure is accomplished by unloading valves, which may be a separate unit or may be mounted on the check valves. The simplest form of relief is provided by a small orifice in the check valve.

Components

Tanks

The tanks are designed to hold the required amount of fuel and are subdivided into main, auxiliary, and external tanks. Depending on the installation, the auxiliary tanks may be located inside the flight vehicle or externally, and the latter may be droppable.

The internal tanks are located in the fuselage or wing. On the modern airplane the fuel weight reaches 35 - 50% of its takeoff weight. In this connection the problems of fuel disposition must be considered in selecting airframe size. Fuel feed from tanks located in the thin tip portions of the wing is poor during maneuvers. Therefore maximal use must be made of the wing root sections. Location of the engines and landing gear nacelles in this region of the wing makes it difficult to locate the fuel there.

The tanks are divided into flexible, rigid, and integral as a function of the tank material and construction.

Flexible tanks are made from fuel-resistant rubber with an outer layer of rubberized fabric (cord), kapron fabric, artificial leather, or waterproof canvas. They either have no structural members or their internal structural elements are not rigidly connected with the tank walls. Since they are located in a container, they can withstand a small differential internal pressure but cannot tolerate negative pressure.

These tanks are widely used. They are easily installed through relatively small hatches and have high survivability. The tanks do not rupture in case of airplane impact with the ground, make the best use of the airframe volume, and are not sensitive to vibration. These tanks are not difficult to fabricate and are lighter in weight than metal tanks of the same volume. They have good thermal insulation qualities and this reduces fuel heating at supersonic flight speeds.

A drawback of the flexible tanks is that they lose their elasticity at low temperatures. This makes their installation difficult under wintertime conditions (the tanks must be preheated using warm air prior to installation) and leads to fuel leakage at points where fittings are installed. Nor is it impossible that some amount of the inner rubber layer may dissolve in the fuel. Therefore the flexible tanks must be flushed out periodically. Their outer layer will be damaged by the action of fuel spilled during refueling.

Rigid tanks are usually fabricated from sheets of the aluminum-magnesium alloys AMn and AMnH (half-hard) and AMnO (annealed). These sheets permit deep drawing, weld well, are elastic and quite resistant to corrosion. The tanks consist of end closures, shells, and an internal structure in the form of baffles to provide tank stiffness and damp out fuel motion.

Usually the end closures and shells are welded together, and the internal structure is riveted in place. There are openings in the baffles to provide free flow between the tank cavities. The rigid tanks are attached to the airframe structure by means of straps and pads to reduce the effect of vibration and deformations of the airplane structure on the tanks. The considerable weight of the rigid tanks, the difficulties involved in installing and removing the tank from the airframe, and the poor resistance to vibration limit their application.

The integral tanks are a variety of the rigid tank. The space and structure of the wing are used to form the fuel tank. Such tanks have good weight characteristics (no special walls or closures) and do not require any special installation and removal operations. Optimal use of the wing volume increases the capacity of the fuel reservoirs. For these reasons the integral tanks are widely used. A drawback is their susceptibility to thermal, aerodynamic, and vibrational loads. Sealing is maintained by the use of sealants which are resistant to heat, cold, and fuel.

The tanks are outfitted with the following equipment to provide for operation of the fuel system: filler necks, sumps, drain cocks, check valves, level limit valves, refueling receptacles, tank interconnections, venting, inerting, metering equipment, mounting flanges, and pumps. Hatches with removable cover plates are provided for tank inspection.

The fuel volume E_f required to fill the tank is

$$E_{\text{т}} = \frac{G_{\text{т}}}{\gamma_{\text{т}}} + E_{\text{н.о}} + E_{\text{исп}},$$

where G_f — is the weight of the fuel required for flight with given maximal range and duration with account for navigational reserve for at least one hour of flight;

γ_f — is the minimal specific weight of the fuel (at 45° C);

E_{un} — is the unusable fuel;

E_{ev} — are the fuel losses by evaporation into the atmosphere.

The unusable fuel quantity arises as a result of the geometric peculiarities of the location of the ABP on or in the tank. The unusable fuel quantity increases when the pumps are located on the tank walls, but must not exceed 1% of the total fuel quantity.

The evaporation losses depend on the type of fuel, the flight time, and the design of the vent line external fittings. For kerosene these losses amount to about 0.2 - 0.5% of the initial fuel quantity per hour of flight.

The total tank volume is

$$E_{б}=E_{т}+\Delta E_{к}+\Delta E_{е},$$

where ΔE_{eq} — is the tank volume occupied by equipment (fuel quantity transmitters, boost pumps, level limit and drain valves, internal structure, and other equipment located inside the tank). We can assume approximately that ΔE_{eq} amounts to 2 - 3 % of the total volume.

ΔE_{ex} — is the tank free volume required for fuel expansion as it heats up (no less than 2% of the total tank volume).

For preliminary calculations we can assume that the total tank volume $E_t = 1.05\ E_f$.

Fuel Lines

The fuel lines serve to connect the components of a given system and transfer the fluid. Depending on the material, they may be rigid or flexible. The lines are subjected to deformations and vibrations as a result of airframe and engine components affecting the lines and also as a result of hydraulic impacts and fuel pressure pulsations. Lines fabricated from rigid tubing must have flexible segments to reduce the vibrational input.

The rigid lines are fabricated from Dural, aluminum-magnesium alloys, brass, and steel. The last is used in high-pressure lines (fuel feed to the nozzles). For corrosion protection, the Al-Mg alloy lines can be anodized and the steel lines can be zinc plated.

Flexible tubes (hoses) are used to connect rigid lines or in areas where installation is difficult. In installing the lines, both high spots where air can accumulate and bends which restrict fuel flow and fuel drainage from the line should be avoided. A small tubing bend radius increases the hydraulic resistance and the stress concentration. It is recommended that bends be made so that the bend radius (along the tubing centerline) will be no less than three times the tubing outside diameter. Elbows are installed at locations where the tubing cannot be bent. Flex hoses, flared lines, and nipple connections are used to connect the rigid lines with one another and with the system components. Flex hoses are connected with one another by means of hose fittings. Tees and crosses are used to interconnect several lines.

The tubing internal diameter is determined by hydraulic and weight analyses of the line. The fluid flow rate must be restricted in order to reduce the hydraulic resistance. The fuel flow velocity in the lines which feed the fuel to the engines is in the range of 2 - 5 m/sec. Higher velocities are possible in the other lines. Although high fuel flow velocities in the lines reduce the line diameter and weight, they also cause large hydraulic resistances and increase the static electricity charge. On the other hand, low fuel flow velocities lead to large line diameters, which increases their weight and creates difficulties in installation.

The wall thickness should not be less than 1 mm for aluminum alloy lines and 0.5 mm for steel lines. The calculated values of the line diameter and wall thickness are adjusted to the dimensions prescribed by GOST 1947-56 for aluminum and aluminum alloy tubing (see Appendix 8) and GOST 8734-58 for seamless cold drawn and cold rolled steel tubing.

Pumps

Hydraulic machines which transfer fluid with the required pressure rise are termed pumps. The pumps used in the fuel systems of aircraft powerplants are divided into vane-type and volumetric pumps.

In the vane pumps the rotating impeller imparts a high pressure to the fluid and forces it into the discharge line with increased velocity. Centrifugal pumps are a type of vane pump.

In the volumetric pumps the fluid is expelled under pressure from the working chamber into the delivery chamber. These pumps are divided into the piston and rotary types. In the rotary pumps the fluid is expelled from the moving working chambers by rotating displacement elements. The rotary vane pumps are of this type.

Usually the ABP and BP are centrifugal; the EBP may be centrifugal or rotary vane types. The advantages of the centrifugal pumps are uniform output, minimal number of moving parts, absence of valves, servicing simplicity, low weight and size, and low cost. The disadvantages include low suction height, moderate efficiency (usually no more than 0.6), and the possibility of flow interruption because of cavitation at low inlet pressures.

The fuel entering the centrifugal pump passes through a screen 1 (Figure 50), which protects the pump against entry of mechanical particles, prevents the formation of a funnel (vortex fuel flow) at the inlet to the pump when there is only a small amount of fuel in the tank, and thus prevents air entrainment through this funnel. After passing through the screen, the fuel is picked up by the axial impeller 2, which creates some head and preswirl of the flow prior to its entry into the centrifugal impeller. The anticavitation qualities of the pump are thus improved, since the axial impeller

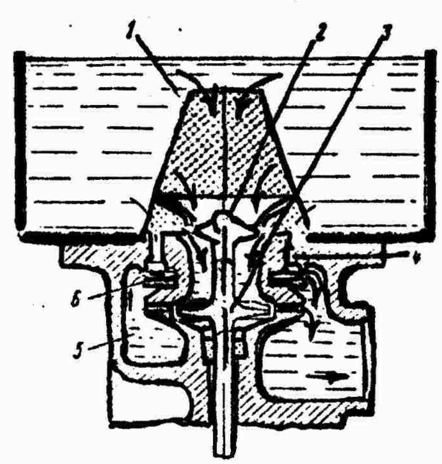


Figure 50. Out-of-tank boost pump (inoperative condition—check valve 6 open)

provides a pressure rise which compensates for the pressure decrease at the inlet to the centrifugal impeller.

The impeller 3 may be of the shrouded, semishrouded, or unshrouded type. The choice of the particular type is determined by the loads acting. The check valve 6 is provided to reduce the hydraulic resistance of an inoperative pump. When the pump is operating the valve covers the port 4 in the inlet part of the pump. If the pump becomes inoperative, the pressure falls off, the valve drops down, and part of the fluid passes through the port, bypassing the impeller. Downstream of the impeller the fuel is directed into the discharge chamber 5, where the kinetic energy of the flow is transformed into pressure energy and the fluid swirl is removed to reduce the resistance. The discharge chamber is usually made in the form of a spiral or annular vaned diffuser. From the diffuser the fluid enters the line.

The centrifugal pump shaft drive may be electrical or mechanical. The latter may be provided from the aircraft engine shaft or by a hydroturbine or pneumoturbine. The electrical drive is lightest for small ABP. Direct current at 27 volts is usually used to supply the electric motors for such pumps. For weight reduction when using more powerful ABP, it is advisable to go to three-phase alternating current at 115/200 Volts and frequency 400 Hz.

The advantage of the alternating current motor is the absence of the commutator and brushes, which increases motor service life and reduces the radio interference level. The ABP with electric drive can be located outside or inside the tank, or in the line.

The external ABP are mounted on the bottom or side walls of the tank. The latter pump location makes better use of the space surrounding the tank but leads to increase of the unusable fuel. The out-of-tank pumps are mounted on the fuel tank so that the pump case with electric motor attached to it is located outside the tank. These pumps are very convenient in operation because of good access to the pump and electric motor. The electric motors are equipped with fans. If they have adequate cooling, the out-of-tank ABP can operate at high outputs without overheating.

The in-tank pumps are mounted on the bottom of the tank (Figure 51). This pump location permits better use of the space around the tank and reduces the unusable fuel. The location of the motor in the fuel, and in some cases fuel circulation around the electric motor case, facilitates motor cooling. However, as the fuel is used from the tank the cooling of the electric motor deteriorates. Therefore extended operation of the in-tank pump without fuel is not permitted. The use of integral tanks has led to the development of submerged, wet-wing pumps. In these pumps the entire pump together with the discharge diffuser and electric power cable are located inside the tank (Figure 52).

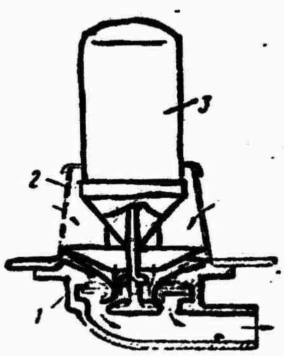


Figure 51. In-tank boost pump:

1 — body; 2 — screen; 3 — electric motor

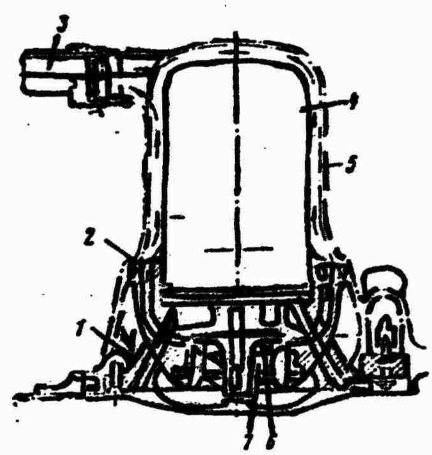


Figure 52. In-tank, wet-wing boost-pump:

1 — body; 2 — annular channel; 3 — delivery line; 4 — electric motor; 5 — housing; 6 — centrifugal impeller; 7 — axial impeller

Location of the boost pumps in the line, far from the tank, has not been widely used. Their ceiling is low because the line segment from the tank to the pump offers hydraulic resistance. Moreover, the air which is released at the pump inlet is not bypassed to the tank, but is directed through the pump to the engine.

When necessary, the ABP may have different rotational speeds. The different pump operating regimes (for example, standby, rated, and overspeed) together with the use of check valves permit programmed fuel feed in the case of parallel connection of groups of tanks in the engine fuel feed system. Increase of the rotational speed of dc electric motors with parallel excitation is achieved by increasing the resistance of the excitation winding.

The standby pump operating regime is obtained with two excitation windings connected in parallel, which corresponds to connection of all the pins of the disconnect plug to the ship's electrical system (Figure 53a). The rated pump operating regime is provided

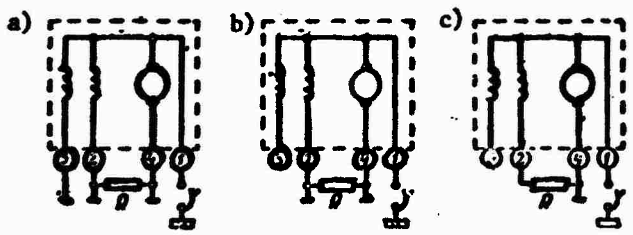


Figure 53. Wiring diagrams of boost pump electric motors for obtaining different regimes:
a — standby; b — rated; c — overspeed

when using a single excitation winding by disconnecting pin 3 (Figure 53b) from the negative wire of the ship's electrical system. The overspeed pump operating regime is obtained by connecting an additional resistor R in the excitation winding circuit, for which pin 2 (Figure 53c) is also disconnected from the negative lead of the ship's electrical system. As this is done the ABP characteristics change (Figure 54).

The centrifugal EBP shaft is driven from the engine shaft. The required rotational speed is obtained by means of a speed reducer. These pumps are equipped with pressure regulators to regulate the pressure with variation of the fuel flowrate.

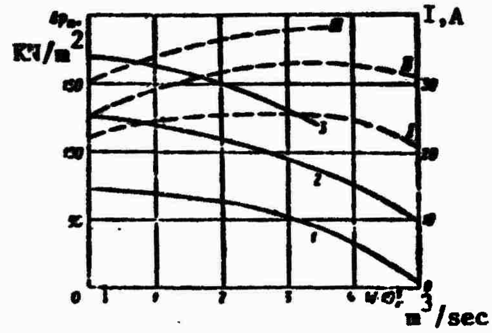


Figure 54. Characteristics of ETsN-T boost pump for different regimes (broken curves are current consumption):

1 — standby; 2 — rated; 3 — overspeed

The ABP shaft can be driven by a pneumoturbine by bleeding air from the engine compressor or by using the dynamic head of the relative airstream. It is advisable to use ABP with air-driven turbine on aircraft with very high fuel flowrates. The weight and size of the pneumatic turbine drive are considerably less than for the electric motor drive. The disadvantage of the ABP pneumatic turbine drive with air bleed from the engine compressor is the fact that the pump cannot provide fuel in the starting and idling regimes. To provide for operation of the ABP with an air-driven turbine under these conditions use can be made of air supplied from bottles, which are then refilled as the engine comes into operation. Fuel may also be supplied under these conditions by an electrically driven pump in parallel with the air-turbine-driven pump.

The air bled from the engine compressor to the ABP pneumatic turbine is at high temperature. Therefore careful sealing of the ABP shaft and thermal isolation of the pump section from the turbine are required. If the sealing and isolation are not sufficient, the fuel temperature will rise. Air bleeding also has an effect on engine operation: the fuel flowrate increases.

The air-turbine-driven ABP using the dynamic pressure of the relative airstream is used in emergency cases and in case of failure of pumps with other types of drive. Such standby pumps are installed in the lower part of the wing. To activate the pump compressed air is supplied to extend the turbine unit into the airstream. Such a drive cannot be used to provide continuous operation of the ABP because of the wing frontal drag increase.

Hydraulic turbine drive of the ABP shaft avoids many of the disadvantages listed above. When fuel is used to drive the turbine, very simple shaft seals can be used and careful thermal isolation of the pump unit from the turbine is not required. Either the EBP or the MEP is used to supply fuel to the ABP turbine. To obtain the required pressure (15 - 25 kgf/cm^2) in the ABP hydraulic drive turbine, a second stage providing suitable pressure must be available in the EBP or bypass of part of the fuel in the MEP must be provided. The advantages of the hydraulic turbine drive are low weight and size, fire safety, simplicity of construction, and reliability in operation. The disadvantage lies in the necessity to supply fuel under starting and idling conditions.

Operation of the ABP with a turbine drive is regulated by varying the flowrate and pressure of the air or fuel supplied to the turbine.

For the centrifugal ABP with turbine drive, the air or liquid flowrate W_t required to drive the turbine can be found from the formula

$$W_{\text{т}} = \frac{\Delta p_{\text{н}}}{\Delta p_{\text{т}}\, \eta_{\text{н}}\, \eta_{\text{т}}} W_{\text{нв}}. \qquad (48)$$

If we take the turbine and pump efficiency to be $\eta_t = \eta_p = 0.7$ and the ratio of the pressure rise created by the pump Δp_p to the pressure differential Δp_t across the turbine to be about 0.05, we find that the fluid flowrate in this case must be about ten times less than the engine fuel flowrate W_{eng}.

Filters

To ensure reliable operation of the engine components, the fuel must not contain mechanical particles. Therefore the fuel is carefully filtered, both in the aircraft fuel system itself and outside this system (prior to fueling).

Filtration is accomplished by passing the fuel through a porous material which retains the mechanical particles whose dimensions exceed the dimensions of the pore cells. Thus, the cell opening dimension characterizes the filtration capability.

Aircraft tanks are fueled with prefiltered fuel. The first filtration is accomplished at the airfield dump delivery station when filling the refueling trucks; the second filtration is accomplished in the fueling process through tanker filters having a filtration capability down to five microns. Servicing of the aircraft tanks with filtered fuel still does not ensure reliable operation of the fuel system components. Particles from the flexible tanks, corrosion products from the lines and components, and bits of rubber can get into the fuel. Therefore the fuel is filtered once again on aircraft with PE and TJE and twice on aircraft with TPE. This is explained by the constructional peculiarities of the fuel equipment of these engines. On the TPE the fuel first passes through a coarse filter with filtration capability down to 100 microns. Then it passes through the fine filter, which retains particles of size greater than 8 - 10 microns.

In order to ensure fuel supply to the engines, the filters are equipped with bypass valves, which open automatically when a given pressure differential across the filtering element is reached if it becomes clogged or freezes over; these bypass valves maintain the required fuel flowrate. With working pressure up to 3 kgf/cm^2, the valve opens when the differential pressure reaches 0.4 - 0.6 kgf/cm^2; with working pressure up to 6 kgf/cm^2, it triggers at 0.7 - 0.9 kgf/cm^2.

The filtering elements are fabricated from screen or paper. The screen material (GOST 6613-53) is nickel, brass, or bronze wire. The screen number indicates the mesh clear dimension (Table 4).

TABLE 4. CHARACTERISTICS OF FILTERING ELEMENTS

Material	AFB-1K paper	Screen					
		Nickel			Bronze	Brass	
		Particle size, mm	Laminated No. 004	No.004	No.0056	No.01	No.025
Cleaning accuracy, mm	8-12	12-16	20-30	40	56	100	280

The screen type filtering elements are made in the form of frames having a cylindrical or star-shaped form, and also in the form of plate-type perforated disks. The paper filtering elements are made from AFB-1K paper, treated with an alcohol solution of bakelite lacquer (grade A). To increase the surface area the paper is gathered into folds which are supported by a metal frame.

Filter selection is based on curves of the hydraulic resistance as a function of the flowrate (see Figure 26).

Fuel Accumulators

Fuel accumulators provide short-term fuel supply to the EBP in the case of nearzero and negative load factors and when the fuel uncovers the ABP during aircraft rolling and sideslipping, and also provide for separation of air from the fuel. They may be designed with a floating piston or a membrane (Figure 55). The latter are encountered more frequently, since they are lighter in weight (they can be made in spherical form).

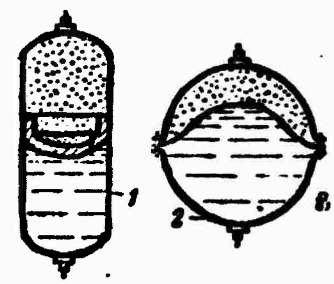


Figure 55. Fuel accumulators: 1 — piston; 2 — membrane

A check valve is located at the fuel inlet into the accumulator. In the case of nearzero or negative load factors, when fuel is not supplied from the tank to the ABP, the check valve closes and the fuel which is in the accumulator fuel chamber is supplied to the EBP under pressure. After termination of the action of these load factors the ABP supplies fuel through the open check valve into the accumulator and to the EBP. The fuel accumulator must be installed close to the ABP and the check valve. As the accumulator is filled with fuel, the air which is released is diverted through the vent line to the tank or to the vent manifold.

Analysis

Basic Data

The design problem includes determination of the system line diameters, selection of the ABP and BP, finding the volume of the fuel accumulators and the ceiling with the ABP operating and not operating, selection of the tank thermal insulation, and strength analysis of the tanks.

In the design analysis the given fuel system operating conditions (fuel flowrate and temperature, flight altitude and speed) are used to find the line diameters and select the ABP and other components which provide the fuel supply to the engines. In the check analysis the available components and lines are used as a basis for making a check: does the given system provide the specified operating conditions. Both forms of analysis are used in designing systems for supplying fuel to the engines.

For the calculation we need to have the following basic data:

schematic of system for supplying fuel to the engines;
design flight regime parameters;
fuel flowrate versus flight altitude;
characteristics of the fuel being used;
head and cavitation curves of the ABP and EBP.

The analysis of systems for supplying fuel to the engines is made for the segments from the tanks to the EBP. The latter (together with components located downstream of them) have already been selected by the engine manufacturer to support the operation of the MEP. The technique presented can also be used to select the EBP.

We analyze the line segments which are in the least favorable conditions for supplying fuel (with respect to line length and relative height of one component above another). Therefore the fuel system schematic must give an idea of the length of the lines and the height of the components. On the basis of the necessity to examine the least favorable conditions, we take the case in which the fuel tank is nearly empty (i.e., the fuel level in the tank can be neglected).

The calculations are made for several regimes. We must check the operation of the fuel feed lines under the most severe operating conditions. Such conditions are the takeoff ground roll and acceleration of the aircraft to the lift-off speed, lift-off and climb at takeoff power, level flight at the design cruising altitude. The load factors are taken from the aerodynamic analysis. If these data are not available, for civil aircraft we can take: $n_y = 4$ and -1. $n_x = \pm 0.3$; $n_z = 0$.

The engine fuel flowrate as a function of flight altitude (Figure 56) is indicated in the engine cahracteristics. The required engine operating regimes are determined in the aerodynamic analysis. In calculating the ceiling of civil aircraft with ABP operating, it is recommended that the nature of the fuel flowrate variation be taken along the curve abcd, so that takeoff power is used on the segment ab and rated power is used on the segment cd, while in calculating the ceiling with the ABP not operating it is recommended that the ceiling be calculated using curve abef, where cruising power is used on the segment ef.

The design fuel temperature for subsonic aircraft is usually 45° C, since in this case the saturated vapor pressure of the fuel has the greatest effect. However, for flight vehicles with long lines it is also necessary to make the calculation for a temperature of minus 60° C, since in this case the hydraulic resistance may be the decisive factor. For supersonic aircraft the design temperature of the fuel in the tanks is selected after analysis of the thermal isolation of the fuel tank walls. Thick thermal insulation layers

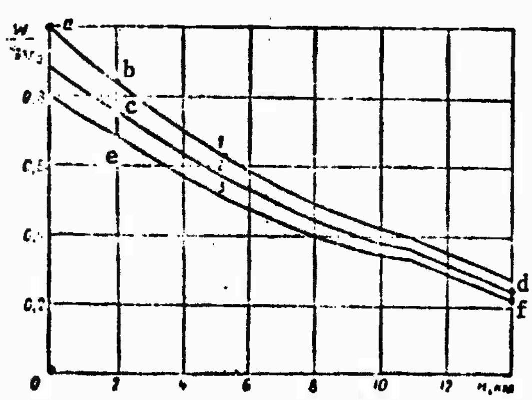


Figure 56. Relative engine fuel flowrate versus flight altitude and power:
1 — takeoff; 2 — rated; 3 — cruise

make it possible to have comparatively low fuel temperature in the tanks, but leads to increased weight. Thin thermal insulation leads to high fuel temperature in the tanks and deterioration of the operation of the boost pumps and the engine feed manifold. Thus, the choice of the design fuel temperature in the tanks of a supersonic airplane is made with account for the weights of the tanks, pumps, and lines, and the power required to drive the pumps.

Boost Pump Selection and Determination of Engine Feed Line Diameter

The selection of the ABP and the determination of the line diameters are made for the maximal fuel flowrate through the engine feed line. For airplanes with gas turbine engines the design altitude is H = 0, and the design flight condition is takeoff.

To obtain the computational formulas we write the Bernoulli equation for the section P-P (Figure 57), located just downstream of the ABP, and the section E-E, at the inlet to the EBP

$$\Delta p_{П} + \frac{\gamma_т V_П^2}{2g} + y_П \gamma_т = p_{вх\,Д} + \frac{\gamma V_Д^2}{2g} + y_Д + \Delta p_г + \Delta p_м, \qquad (49)$$

where Δp_P — is the pressure rise across the ABP;

V_P and V_E — are the fuel velocities at P-P and E-E;

y_P and y_E — are the heights of sections P-P and E-E;

p_{inE} — is the EBP inlet pressure.

In (49) all the terms represent pressure expressed in Newtons per square meter.

The minimal pressure rise which the ABP must develop is

$$\Delta p_{П\,\min} = p_{вх\,Д\,\min} + \gamma_т \frac{V_Д^2 - V_П^2}{2g} - (y_П - y_Д)\gamma_т + \Delta p_г + \Delta p_м. \qquad (50)$$

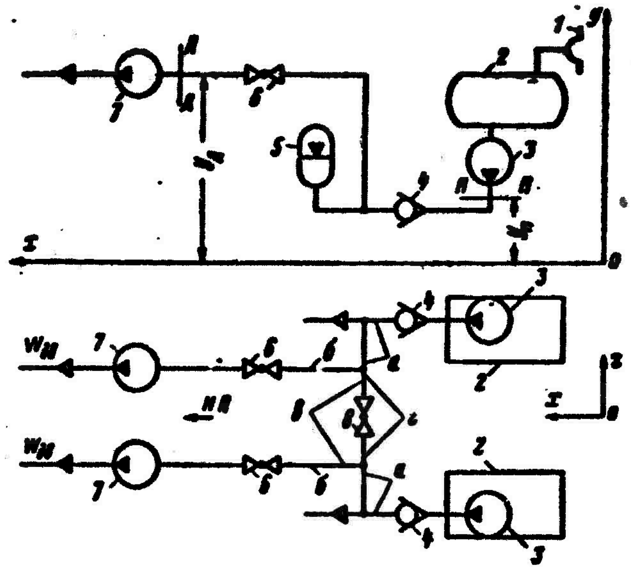


Figure 57. Computational diagram for engine fuel feed lines;

1 — air intake from atmosphere; 2 — tank; 3 — airframe boost pump; 4 — check valve; 5 — fuel accumulator; 6, 8 — fire shutoff and crossfeed valves; 7 — engine boost pump

To ensure cavitation-free operation of the EBP, it is necessary that

$$p_{\text{вх Д}\,\min} = p_{t\,4/1} + \Delta p_{\text{кав ПНД}\,\min}. \tag{51}$$

Substituting (51) into (50) and replacing Δp_h by its value from (24), we obtain

$$\Delta p_{\text{П}\,\min} = p_{t\,4/1} + \Delta p_{\text{кав ПНД}\,\min} + \gamma_{\text{т}} \frac{V_{\text{Д}}^2 - V_{\text{П}}^2}{2g} - (y_{\text{П}} - y_{\text{Д}})\gamma_{\text{т}} + c^2 k_{\text{пот}}^2 \gamma_{\text{т}} W_{\text{расч}}^2 + \Delta p_{\text{н}}. \tag{52}$$

If the line internal diameter is constant along the length of the engine feed line, the fuel flow velocities are also constant. Then the difference of the dynamic pressures $\gamma_f \dfrac{V_E^2 - V_P^2}{2g} = 0$. For different values of the line internal diameter, we use the condition of a constant value of the fuel velocity through all the lines. Then (52) can be written in the form

$$\Delta p_{\text{п min}} = p_{t\,4/1} + \Delta p_{\text{r.з ПНД min}} - (y_{\text{п}} - y_{\text{Д}})\gamma_{\text{т}} + \Delta p_{\text{н}} + $$
$$+ c^2 \gamma_{\text{т}} W^2_{\text{расч}} k^2_{\text{маг}} = A + B k^2_{\text{маг}}, \tag{53}$$

where A — is a quantity which is independent of the line diameter;

B — is the coefficient of K^2_{line}.

These coefficients are found from the formulas

$$A = p_{t\,4/1} + \Delta p_{\text{r.з ПНД min}} - (y_{\text{п}} - y_{\text{Д}})\gamma_{\text{т}} + \Delta p_{\text{н}}; \tag{54}$$

$$B = c^2 \gamma_{\text{т}} W^2_{\text{расч}}. \tag{55}$$

In calculating the reduced hydraulic resistance coefficient of the line elements using (26), we must know the line diameter. To find the latter we must know for the given fuel flowrate the value of the fuel flow velocity, which in turn depends on the line diameter. Then we assume values of the velocity on the basis of statistical data in order to obtain the solution. A more exact solution can be obtained if we use successive approximations of a grapho-analytical technique.

If there are several engines and service tanks with ABP, various versions of the fuel flow through the engine feel line and, correspondingly, different values of the reduced hydraulic resistance coefficient k_{line} are possible. By examining three possible fuel feed versions: single ABP to two engines (with crossfeed valve open), single ABP to a single engine (with crossfeed valve closed), and two ABP to a single engine (with the crossfeed valve open), we can establish that the first version is the design condition.

<u>Fuel supply from a single pump to two engines</u>. If there is a failure of the ABP of a single independent engine fuel feed line,

it is necessary to supply fuel to the engine from the pump of the other independent line and open the crossfeed valve. In place of independent lines supplying fuel to the engines we obtain centralized fuel supply by a single ABP to two engines along a branched line which includes the crossfeed line. Thus, in finding the diameter of the engine feed manifold lines we also determine the diameters of the piping in the crossfeed line. While prior to opening the crossfeed valve the maximal fuel flowrate through the ABP was W_{eng}, it will now be $W_{des} = 2W_{eng}$.

In those cases in which the ABP supplies not only the fuel supply to the EBP but also cooling of the oil in the fuel/oil radiator, operation of the hydraulic turbine drive of the ABP itself, opening and closing the propulsive nozzle eyelids, rotation of the compressor inlet guide vanes, filling the accumulator fuel chamber (the flowrate through the ABP is more than the fuel flowrate required by the engines and part of the fuel is bypassed back to the tank), it is necessary to make the calculation using the increased fuel flowrate W_e

$$W_{д}=W_{дв}+W_{пер}+W_{а.з},$$

where W_{by} — is the flowrate in the bypass lines;

$W_{a.f.}$ — is the fuel flowrate to fill the accumulator fuel chamber.

When the crossfeed valve is opened in case of failure of the ABP of a single independent line and fuel is crossfed, the design value of the fuel flowrate is

$$W_{расч}=2W_{д}=2(W_{дв}+W_{пер}+W_{а.з}).$$

Let us examine the graphoanalytical solution of the problem of determining the engine feed line diameters and selecting the ABP. We take several values of the line internal diameter d_a on the segment a (see Figure 57). This makes it possible to use (16) to find the Reynolds number. Then for the turbulent flow regime, which is usually encountered, we can use (18), (19), (20) to find the frictional resistance coefficient λ. Knowing the length of the segment a and the local resistance coefficients for this segment, we can use (23) to find the equivalent hydraulic resistance coefficient ζ_{eq}. We calculate the reduced hydraulic resistance coefficient k_a for segment a using (26). Taking the value of the fuel velocity V to be constant through all the lines, we write

$$V=\frac{4W_a}{\pi d_a^2}=\frac{4W_b}{\pi d_b^2}=\frac{4W_e}{\pi d_e^2}=\frac{4W_i}{\pi d_i^2}. \qquad (56)$$

Then

$$d_i=d_a\sqrt{\frac{W_i}{W_a}}. \qquad (57)$$

where $W_a = W_{des} = 2W_e$.

The relation (57) makes it possible to find the values of k_b, k_c and then use (35) to find k_{line}. Since the fuel accumulators are usually located near the engine feed line, the hydraulic resistance of the line leading to the accumulator can be neglected and we need consider only the local branching resistance. If the fuel accumulator is installed far from the feed line, this branch line must be considered.

Using (53), we can plot a graph of the variation of the minimal pressure rise $\Delta p_{P\,min}$ which the ABP must develop as a function of the diameter d_a (curve 4 in Figure 58). We need one more equation to solve the problem with these two unknowns. The ABP head curve can be used to represent the missing equation in graphical form.

Knowing the fuel flowrate W_{des}, we can use the head curve of the selected ABP (curve 2) to determine the pressure rise created by the boost pump, which is equal to the minimal pressure differential required for the feed line, and the corresponding diameter d_a. Using (57), we obtain the diameters of the other line segments. The resulting calculated line diameters are adjusted to fit the standard line sizes available (Appendix 8).

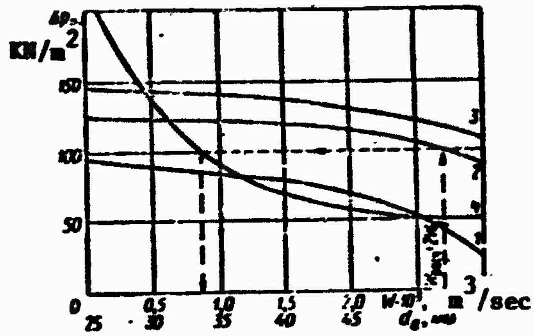


Figure 58. Airframe boost pump selection and determination of line diameter:

1, 2, 3 — ABP head curves (ETsN-45, ETsN-T and ETsN-10 at rated power operation; 4 — minimal required ABP pressure

If several types of ABP are available, different versions of the choice of the required ABP pressure rise and line diameter are possible. Here we must bear in mind that large line diameters lead to increase of the piping weight and small diameters require large ABP pressure rises and correspondingly high power to drive the ABP shaft. An integrated analysis of all these factors makes it possible to arrive at the optimal solution for the given conditions. When the head characteristics of the available pumps do not satisfy the required flowrate and pressure values, a new pump must be ordered or the available pump must be modified (see Chapter II).

After selecting the ABP and determining all the line diameters, we can plot the manifold characteristic curve (curve 2 in Figure 59).

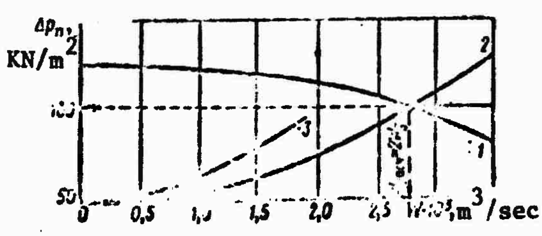


Figure 59. ABP head curves and characteristics of engine feed manifolds with crossfeed valve open and closed:

1 — ABP head curve; 2 — crossfeed valve open (centralized supply to two engines; 3 — crossfeed valve closed (independent feed)

Then its intersection with the head curve of the selected ABP (curve 1) corresponds to the design fuel flowrate value $2W_e$. With the crossfeed valve closed (curve 3) and the fuel flowrate W_e, there will be additional excess pressure at the inlet to the EBP.

Transfer Pump Selection and Determination of Transfer Manifold Line Diameter

The calculation is made for the maximal fuel transfer rate from Tank 5 (Figure 60), equal to the maximal flowrate from the service tank 1. It is clear that in this case the maximal flowrate will occur with the crossfeed valve open in case of failure of the ABP of a single independent line. For airplanes with gas turbine engines the design altitude H = 0, and the takeoff flight regime is the design condition.

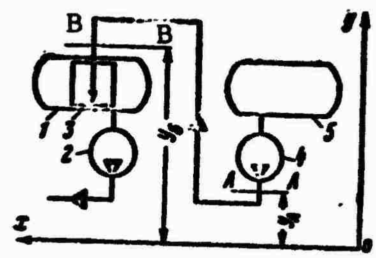


Figure 60. Computational diagram for fuel transfer manifold: 1 — service tank; 2, 4,— boost and transfer pumps; 3 — safety float valve; 5 — tank

We write the Bernoulli equation for the section A-A, located just downstream of the BP of tank 5, and the section B-B at the inlet to tank 1

$$\Delta p_{\text{пн}} + \frac{\gamma_т V_A^2}{2g} + y_A \gamma_т = p_{\text{вх}_Б} + \frac{\gamma_т V_Б^2}{2g} + y_Б \gamma_т + \Delta p_r + \Delta p_м,$$

where p_{inB} — is the pressure at the tank inlet.

The minimal pressure rise which the BP must develop is

$$\Delta p_{\text{пн}\,\min} = p_{\text{вх}_{Б\min}} + (y_Б - y_A)\gamma_т + \Delta p_r + \Delta p_м. \qquad (58)$$

To ensure fuel transfer the minimal pressure at the inlet to tank 1 must be

$$p_{\text{вх}_{\Gamma\min}} \ll \frac{P}{F_{\text{к}}}. \tag{59}$$

where P — is the closure force acting on the valve;

F_v — is the valve area.

Therefore we can take $p_{\text{in}_{\text{Bmin}}} = 0$.

Substituting (59) into (58) and replacing Δp_h by its value from (24), we obtain

$$\Delta p_{\text{ПН}\min} = (y_{\text{Б}} - y_{\text{А}})\gamma_{\text{т}} + \Delta p_{\text{н}} + c^2 k^2_{\text{маг}} \gamma_{\text{т}} W^2_{\text{расч}} = A_1 + B_1 k^2_{\text{маг}}, \tag{60}$$

where

$$A_1 = (y_{\text{Б}} - y_{\text{А}})\gamma_{\text{т}} + \Delta p_{\text{н}}; \tag{61}$$

$$B_1 = c^2 \gamma_{\text{т}} W^2_{\text{расч}}. \tag{62}$$

Selection of the transfer pump and determination of the transfer manifold line diameter are also accomplished graphoanalytically (see the preceding analysis). The calculation is facilitated by the fact that the transfer manifold is usually a simple line. The required minimal BP pressure rise is less than the ABP pressure rise.

For the valve to be in the closed position, it is necessary that

$$G_{\text{т}} i > \Delta p_{\text{от}} F_{\text{к}},$$

where G_f — is the weight of the fuel in the float volume;

i — is the gear ratio of the lever arms from the float to the valve;

Δp_{op} — is the differential pressure on the valve required for opening.

The maximal value of the differential pressure will occur for W = 0. Then

$$\Delta p_{\text{от max}} = \Delta p_{\text{ПН}_{W=0}} - A_1.$$

The minimal value of the fuel weight in the float volume

$$G_{\text{т min}} = \frac{\Delta p_{\text{от max}} F_{\text{к}}}{i} = \frac{\Delta p_{\text{ПН}_{W=0}} - A_1}{i} F_{\text{к}}.$$

The minimal float volume

$$E_{\min} = \frac{G_{\text{т min}}}{\gamma_{\text{т}}} = \frac{\Delta p_{\text{ПН}_{W=0}} - A_1}{i\gamma_{\text{т}}} F_{\text{к}}. \qquad (63)$$

Determining Diameter of Gravity Transfer Manifold Line

The gravity transfer manifold (Figure 61) makes it possible to accomplish fuel flow by gravity. This line can be considered a transfer manifold without a pump. Then (60) can be written in the form

$$(y_{\text{А}} - y_{\text{Б}})\gamma_{\text{т}} = \Delta p_{\text{и}} + c^2 k^2 \gamma_{\text{т}} W^2.$$

The difference h_t of levels is required for fuel gravity transfer

$$h_{\text{т}} = y_{\text{А}} - y_{\text{Б}} = \frac{\Delta p_{\text{и}} + c^2 k^2 \gamma_{\text{т}} W^2}{\gamma_{\text{т}}}.$$

If we neglect the magnitude Δp_i of the inertial pressure losses, the required fuel column height is

$$h_{\text{т}} = c^2 k^2 W^2 = c^2 \frac{\zeta_0}{d^4} W^2. \qquad (64)$$

Assuming values of the gravity transfer manifold line diameter and the flowrate through the line, we can for a specific manifold (for definite values of line length and local resistances) use (64)

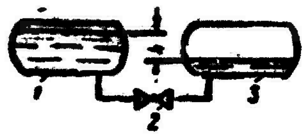


Figure 61. Computational diagram for gravity transfer line:
1, 3 — tanks; 2 — tank interconnect valve

to plot curves of manifold diameter as a function of flowrate and fuel column height (Figure 62).

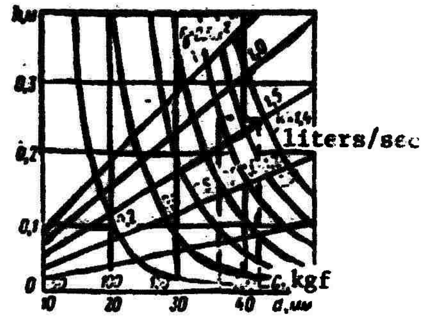


Figure 62. Determining gravity transfer line diameter (example)

In order to obtain weight relationships, we assume that gravity fuel transfer must occur at a definite fuel weight G_f in one of the tanks. Then the volume of this fuel is

$$E_т=\frac{G_т}{\gamma_т}.$$

The volume E_f is in turn connected with the tank area F_t and the fuel column height by the relation $E_f = F_t h_f$.

The available fuel head at which transfer must begin is

$$h_т=\frac{E_т}{F_б}=\frac{G_т}{\gamma_т F_б},$$

hence

$$G_т=\gamma_т F_б h_т=\gamma_т F_б c^3\frac{\zeta_б}{d^4}W_т^2. \qquad (65)$$

Figure 62 also shows curves of the fuel weight at which gravity transfer begins as a function of the fuel column height for a definite fuel grade and different tank area.

Specifying the fuel weight at which its gravity transfer must begin and knowing the tank area, we can find the fuel column height in the given tank. From this value and the required flowrate, we find the required gravity transfer line diameter.

Determining Fuel Accumulator Volume

The total volume of the fuel accumulator is made up of the volumes of the fuel and air chambers. The accumulator fuel chamber volume is found from the condition of short-term provision of fuel flow from the accumulator to the engine with the flowrate W_{acc}, equal to the engine fuel flowrate plus the flowrate through the bypass manifolds W'_{eng}

$$W_{\text{а.р}} = W'_{\text{дв}} = \frac{E_{\text{п}}}{\tau_{\text{р}}} \text{ m}^3/\text{sec}$$

where E_{use} — is the useful volume of the accumulator fuel chamber, m^3;

τ_{dis} — is the time during which the required fuel flowrate is provided, sec.

Hence

$$E_{\text{п}} = W'_{\text{дв}} \tau_{\text{р}} \text{ м}^3.$$

On the basis of the specified time for providing the flowrate W'_{eng} from the accumulator, we can find the useful volume of its fuel chamber.

After depletion of the fuel from the accumulator, the ABP must supply fuel flow to the engine, fuel bypass, and filling of the accumulator fuel chamber

$$W_{\text{н}} = W'_{\text{дв}} + W_{\text{а.з}} = W'_{\text{дв}} + \frac{E_{\text{п}}}{\tau_{\text{з}}},$$

where W_{af} and τ_{fill} — are the fuel flowrate to fill the accumulator chamber and the time required for filling.

Then

$$\tau_з = \frac{E_п}{W_н - W'_{хп}}.$$

The air pressure p_a supplied to the accumulator air chamber must be somewhat less than the fuel pressure p_f. When the air pressure $p_a > p_f$, fuel will feed from the accumulator.

To find the total accumulator volume E_0, we express the useful accumulator volume (Figure 63) in the form

$$E_п = E_1 - E_2 \qquad (66)$$

where E_1 and E_2 — are the accumulator volumes for the air pressures p_1 at the initiation of fuel filling and p_2 at the end of filling.

Assuming that the air expansion and compression process is isothermal, we can write the relation $E_1p_1 = E_2p_2 = E_0p_0 = Ep$; then

$$E_1 = \frac{E_0 p_0}{p_1}, \quad E_2 = \frac{E_0 p_0}{p_2}.$$

and therefore

$$\frac{E_1 - E_2}{E_0} = \frac{E_п}{E_0} = \frac{p_0}{p_2}\left(\frac{p_2}{p_1} - 1\right),$$

where p_0 — is the accumulator pressure when it is charged with air only.

Usually $p_0 = p_1$. Then we obtain

$$\frac{E_п}{E_0} = 1 - \frac{p_1}{p_2}.$$

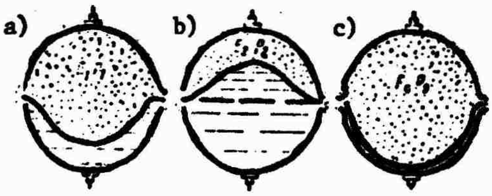


Figure 63. Fuel accumulator operation:

a — initiation of fuel filling; b — end of filling; c — fuel expulsion when its pressure decreases

The pressure p_2 is somewhat less than that developed by the ABP because of hydraulic losses in the segment from the pump to the accumulator.

To determine p_1, we write the elementary energy (work) dA in the form

$$dA = pdE = E_0 p_0 \frac{dE}{E}.$$

Integrating in the limits from E_2 to E_1, we obtain

$$A = E_0 p_0 \ln \frac{E_1}{E_2} = E_0 p_0 \ln \frac{p_2}{p_1}.$$

for $p_0 = p_1$

$$A = E_0 p_1 \ln \frac{p_2}{p_1}.$$

Assuming that the pressure p_2 and total accumulator volume E_0 are given, we can find the optimal pressure ratio $\frac{p_2}{p_1}$, which will correspond to maximal energy storage. To do this we equate the derivative to zero

$$\frac{dA}{dp_1} = E_0 \left(\ln \frac{p_2}{p_1} - 1 \right) = 0.$$

Then

$$\ln \frac{p_2}{p_1} = 1$$

Consequently

$$\left(\frac{p_2}{p_1} \right)_{\text{опт}} = \left(\frac{E_1}{E_2} \right)_{\text{опт}} = e = 2,72.$$

Using this relation we find

$$E_2 = \frac{E_1}{2,72} = 0,367 E_1.$$

Expression (66) can now be rewritten as

$$E_{и} = E_1 - E_2 = E_1 - 0{,}367E_1 = 0{,}633E_1.$$

Taking $E_1 = E_0$, we obtain $E_{use} = 0, 663\ E_0$. Therefore

$$E_0 = \frac{E_{и}}{0{,}633} = 1{,}58E_{и}. \qquad (67)$$

Ceiling of Fuel System with Inoperative ABP

The fuel system ceiling is the highest aircraft flight altitude to which the required uninterrupted fuel supply to the engines is provided. High flight altitude affects the operation of the fuel system components because of cavitation phenomena. The use of ABP makes it possible to obtain sufficient pressure at the inlet to the EBP and the required fuel system ceiling, but when the ABP fails (is turned off) the ceiling will be low.

Uninterrupted fuel supply to the engines with the ABP inoperative must be provided for subsonic aircraft with the engines operating from idle to takeoff power at flight altitudes up to 2000 m and at idle power up to the cruising altitude of 8000 m.

Let us examine a very simple fuel system in the form of an engine feed manifold with inoperative ABP (Figure 64). We write the Bernoulli equation for sections T-T and E-E

$$p_H + \Delta p_{изб} + y_З \gamma_т + \frac{\gamma_т V_Б^2}{2g} = p_{вх\,Д} + y_Д \gamma_т + \frac{\gamma_т V_Д^2}{2g} + \Delta p_г + \Delta p_м, \qquad (68)$$

where p_H — is atmospheric pressure;

V_T — is the rate of lowering of the fuel level in the tank, which can be neglected.

Solving (68) for p_H and using the fact that the maximal flight altitude corresponds to the minimal value of the pressure $p_{H_{min}}$, which will occur at the minimal pressure required $p_{in_{Emin}}$, other conditions being the same, we obtain

$$p_{H\,min}=p_{\text{вх}_{\text{Д}}\,min}+\Delta p_{r}+\frac{\gamma_{\text{т}}V_{\text{Д}}^{2}}{2g}+\Delta p_{\text{и}}-(y_{\text{Б}}-y_{\text{Д}})\gamma_{\text{т}}-\Delta p_{\text{н.б.}} \qquad (69)$$

In (69) all terms are pressures expressed in Newtons per square meter.

Knowing $p_{H_{min}}$, from standard atmosphere tables (Appendix 2) we can find the fuel system ceiling. This solution of the problem corresponds to the check calculation. In the design calculation the required flight altitude and the corresponding p_H are known. Then (69) can be solved for the required pressure at the EBP inlet.

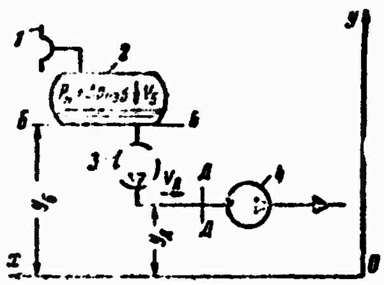


Figure 64. Computational diagram for determining ceiling of fuel system with ABP inoperative:

1 — air intake from atmosphere; 2 — tank; 3 — inoperative ABP; 4 — EBP

We can see from (69) that the following factors affect the ceiling of the fuel system with inoperative ABP:

minimal pressure required at the EBP inlet;

hydraulic resistances;

inertial pressure losses;

relative positioning of the tank and the EBP;

excess pressure in the tank.

The minimal pressure required at the EBP inlet is determined by the condition of cavitation-free operation using (40) and depends on the saturated vapor pressure $p_{t\ 4/1}$ and the minimal required cavitation pressure margin $\Delta p_{cav_{min}}$. The use of fuels with high saturated vapor pressures and the choice of EBP with high required cavitation pressure margin lead to a low ceiling. Therefore, to reduce the pressure required at the EBP inlet it is desirable to have a fuel with low saturated vapor pressure, prevent fuel heating in the tanks and lines, and choose an EBP with low required cavitation pressure margin.

At quite large fuel flow velocities the hydraulic resistances reduce the ceiling. They are found using (24). When the ABP are inoperative we must include in the magnitude of the line hydraulic resistances the additional resistance which arises as the fuel flows through the ABP

$$\Delta p_{\text{г}_{\text{пнл}}} = aW^2.$$

The value of the coefficient a depends on the construction of the ABP. For preliminary calculations we can assume that a = $1.2 \cdot 10^9$ N $\cdot$ sec^2/m^3.

The following measures are recommended to reduce the hydraulic resistances: use fuels with low viscosity, install as few components as possible in the feed manifold segment from the tank to the EBP, run lines of minimal length and large bend radii, use ABP with an annular inlet check valve, which reduces the hydraulic resistance of the inoperative ABP.

In calculating the ceiling, the inertial pressure losses are not taken into account. We consider steady flight without acceleration. If the ABP are not operating, in accordance with the flight operating manual the crew must fly the airplane without developing accelerations.

In modern aircraft the relative positioning of the tank and the EBP has very little effect on the fuel system ceiling. The aircraft configurations lead to relatively small vertical distances between the tank and the EBP.

<u>Excess pressure</u> in the tank leads to increase of the fuel system ceiling. The permissible magnitude of the excess pressure is limited by tank strength and for subsonic flight vehicles usually does not exceed 30 KN/m^2.

To find the ceiling using (69) it is more convenient to write it in the form

$$p_{H\,\min}=p_{t\,ч.л}+\Delta p_{кав\,Д\,\min}-(y_Б-y_Д)\gamma_т-\Delta p_{изб}+(c^2k_c^2\gamma_т+a)W^2= \\ =A_2+B_2W^2. \qquad (70)$$

where A_2 — is a quantity which is independent of the fuel flowrate;

B_2 — is the coefficient of W^2, which depends on the fuel flowrate.

The quantities A_2 and B_2 are found from the formulas

$$A_2=p_{t\,ч.л}+\Delta p_{кав\,Д\,\min}-(y_Б-y_Д)\gamma_т-\Delta p_{изб}; \qquad (71)$$

$$B_2=c^2k_c^2\gamma_т+a. \qquad (72)$$

Since (70) takes into account the required minimal value of the pressure at the EBP inlet with regard to the condition for cavitation-free operation, it can be called the equation of permissible fuel flowrate for cavitation-free operation with inoperative ABP.

In (70) there are two unknowns: $p_{H_{\min}}$ and W. The problem is solved graphoanalytically using the second equation in the form of the required fuel flowrate as a function of the flight altitude. We assume in the present case that the fuel flowrate corresponds to the takeoff flight regime from sea level to 2000 m, and to the cruising regime above 2000 m (see Figure 56, curve abef). In Figure 65, curve 1-2 is the variation of the required fuel flowrate

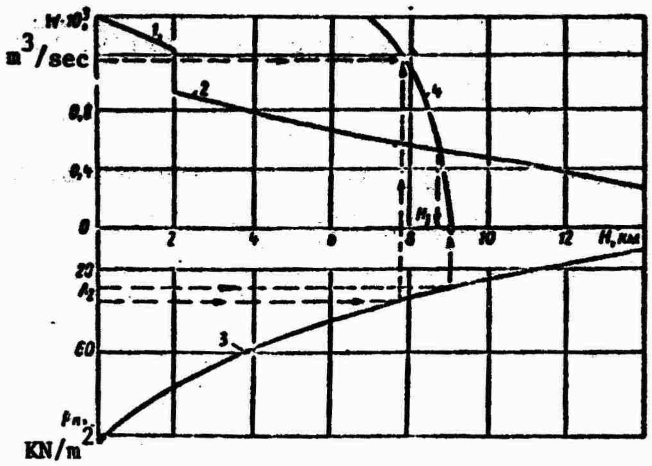


Figure 65. Determining ceiling of fuel system with ABP inoperative

W as a function of flight altitude when the fuel is supplied through an independent engine feed manifold. Curve 4 corresponds to (70) and shows the permissible fuel flowrate W_{per} on the basis of cavitation-free operation of the fuel system. The operating point, located at the intersection of curves 1-2 and 4, indicates the fuel system ceiling H_1.

The calculation sequence is analogous to that used above and amounts to the following. We take several values of the fuel flowrate in the region of intermediate flight altitudes and determine the Reynolds numbers; then we find the frictional resistance coefficient λ, and equivalent ζ and reduced k hydraulic resistance coefficients. Assuming initially that $\Delta p_{ex} = 0$, we use (71) to calculate the value of A_2 and use (70) to find the values of p_{Hmin} for several assumed values of W'_{eng}.

Using standard atmosphere tables or curve 3, which relates air pressure and flight altitude in the standard atmosphere, we plot curve 4 of the permissible fuel flowrate for cavitation-free operation of the EBP.

If the ceiling H_1 is lower than 8000 m, excess pressure must be provided in the tank. The magnitude of the required excess pressure is

$$\Delta p_{изб} = p_{H_1} - p_{8000},$$

where p_{H_1} and p_{8000} — are the air pressures in the standard atmosphere at the altitudes H_1 and 8000 m.

Ceiling of Fuel System with ABP Operating

The determination of the ceiling of the fuel system with ABP operating reduces to determining the ceiling of the ABP themselves.

We write the Bernoulli equation relative to the fuel level in the tank, section T-T, and the ABP inlet, section P-P (Figure 66)

$$p_H + \Delta p_{изб} + y_Б \gamma_т + \frac{\gamma_т V_Б^2}{2g} = p_{вхП} + y_П \gamma_т + \Delta p_т + \frac{\gamma_т V_П^2}{2g} + \Delta p_и. \qquad (73)$$

We assume the fuel flow velocity V_T in the tank to be zero. We also neglect the hydraulic resistances and inertial losses. Then the minimal value of p_H is defined by the formula

$$p_{H\,min} = p_{вхП\,min} - (y_Б - y_П)\gamma_т - \Delta p_{изб}.$$

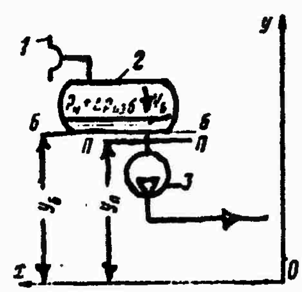


Figure 66. Computational diagram for determining ceiling of fuel system with ABP operating:

1 — air intake from atmosphere; 2 — tank; 3 — ABP

Neglecting the quantity $(y_T - y_P)\gamma_f$ and using (40) for cavitation-free pump operation, we can write (73) in the form

$$p_{H\,min} = p_{tП} + \Delta p_{кавП\,min} - \Delta p_{изб}. \qquad (74)$$

From analysis of (74) we can find that the saturated vapor pressure of the fuel, the magnitude of the excess pressure in the tank air space, and the maximal required boost pump cavitation pressure margin affect the ceiling of the fuel system with operating ABP. In determining the cavitation pressure margin required we must consider that the aircraft rate of climb also affects cavitation in the fuel systems.

The weight of the air dissolved in the fuel is proportional to the air pressure above the fuel (Henry's law). During climb the weight of the air dissolved in the fuel decreases in proportion to the pressure reduction in the fuel tank. The excess air is released in the form of bubbles, which gradually leave the fuel. If the aircraft rate of climb is low the air pressure rate of decrease is also small and the air escaping from the fuel has no marked effect on the operation of the ABP and the fuel system feed manifold.

High aircraft rate of climb, particularly at high flight altitudes, and the marked reduction of the air pressure lead to large amounts of excess air in the fuel, which is not able to escape from the fuel. The result is transient boiling of the fuel. In this case the fuel entering the ABP is supersaturated with air, which leads to vortex formation, earlier onset and development of cavitation in the pump, and entry of air into the feed manifold. The latter may create fuel pressure fluctuations associated with the peculiarities of two-phase liquid flow. Therefore, in the case of large vertical speeds (V_y > 10 m/sec) at high altitudes it is necessary to increase the value of the ABP minimal cavitation margin by the magnitude δp_{cav_P} and determine it from the formula

$$\Delta p'_{\text{кав}\Pi\,\min} = \Delta p_{\text{кав}\Pi\,\min} + \delta p_{\text{кав}\Pi}. \qquad (75)$$

For centrifugal ABP according to (43)

$$\Delta p_{\text{кав}\Pi\,\min} = \frac{10\,(60n)^{4/3}\,W^{2/3}\,\gamma_{\text{т}}}{C^{4/3}} \quad \text{N/m}^2$$

where n — is the pump speed in rps.

It has been established for these pumps that

$$\delta p_{\text{кавП}} = 48 k_p (V_y - 10) \lg V_y \ \text{N/m}^2$$

where k_p — is a coefficient account for the saturated vapor pressure of the fuel.

This coefficient is defined by the formula

$$k_p = 1 + 23 \cdot 10^{-6} (p_{t\,4/1} - 4000). \tag{76}$$

In (76) the values of $p_{t\ 4/1}$ are given in Newtons per square meter.

Then (75) can be written as

$$\Delta p'_{\text{насП}_{\min}} = \frac{10 (60\pi)^{4/3} W^{2/3} \gamma_T}{C^{4/3}} + 48 k_p (V_y - 10) \lg V_y \ \text{N/m}^2. \tag{77}$$

If the fuel system ceiling is higher than the aircraft aerodynamic ceiling, then $V_y = 0$ and $\delta p_{cav_P} = 0$. Therefore the ceiling calculation should initially be made without account for V_y. If as a result of these calculations the fuel system ceiling is lower than the aerodynamic ceiling, then we must determine δp_{cav_P}. The values of V_y are taken from the aerodynamic analysis. To simplify the calculations we can use constant values of V_y at the maximal possible flight altitude, which was determined earlier without account for V_y.

Replacing (77) with (74), we obtain the values of the ceiling with account for V_y

$$p_{H\min} = p_{t\,4/1} + 48 k_p (V_y - 10) \lg V_y - \Delta p_{\text{изб}} + \frac{10 (60\pi)^{4/3} W^{2/3} \gamma_T}{C^{4/3}} = A_3 + B_3 W^{2/3} \ \text{N/m}^2, \tag{78}$$

where

$$A_3 = p_{t\,4/1} + 48k_p(V_a - 10)\lg V_a - \Delta p_{вх\,6} \ \text{N/m}^2; \tag{79}$$

$$B_3 = \frac{10(60n)^{4/3}\gamma_T}{C^{4/3}}. \tag{80}$$

Formula (78) is the equation of the permissible fuel flowrate for cavitation-free operation of the fuel system with ABP operating.

In (78) there are two unknowns: p_{Hmin} and W; therefore the problem is solved graphoanalytically. We assume that the fuel flowrate at altitudes from sea level to 2000 m corresponds to takeoff power and above this altitude to rated power (curve 1 in Figure 67). If we plot on the figure the curve of permissible fuel flowrate for cavitation-free operation of the fuel system with operating ABP (curve 4), its intersection with curve 1 indicates the ceiling H_2.

The calculation sequence is the same as in calculating the ceiling with inoperative ABP.

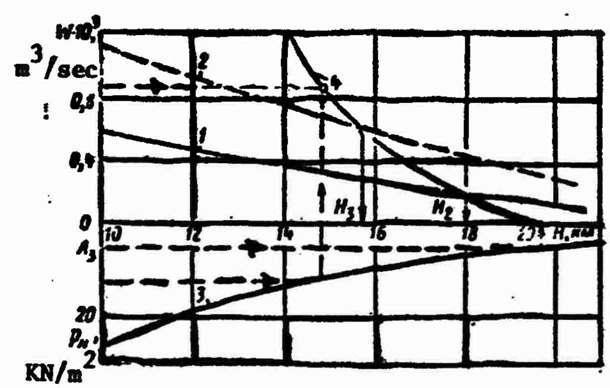


Figure 67. Determining ceiling of fuel system with ABP operating: 1 — required engine fuel flowrate at rated power with crossfeed valve closed; 2 — same, with crossfeed valve open; 3 — air pressure in standard atmosphere; 4 — permissible fuel flowrate for cavitation-free operation of ABP

To determine the ceiling of the fuel system in the case when the crossfeed valve is open, after failure of the ABP of a single independent feed line, we draw curve 2 corresponding to the increased fuel flowrate. The new design indicates the fuel ceiling H_3, which will be lower than the ceiling H_2.

Selection of Tank Thermal Insulation and Tank Strength Analysis

Flight of a supersonic airplane is accompanied by increase of the temperature of its surfaces and the fuel as a result of aerodynamic heating (in flight), radiation of the sun, earth, and atmosphere, heat influx from the operating engines, electronic and other equipment, and also the temperature rise of the fuel in the pumps and as it flows through the fuel/oil radiators. When the fuel and the fuel system components are heated some of the thermal energy is transformed into radiant energy, and this propagates into the surrounding space. The rest of the thermal energy goes to increase the heat content of the fuel and thus increases its temperature.

When fuel is stored aboard aircraft, the maximal temperature rise is observed in the summertime in the southern regions of the USSR, where the air temperature may reach 45° C. If we neglect the thermal insulation effect of the aircraft skin and tank walls, the maximal design temperature of the fuel at sea level can be taken as 45° C.

As the fuel temperature is increased it oxidizes intensely, and as a result there are formed insoluble deposits and resins which clog the filtering elements and disrupt normal operations of the engine fuel system. For example, deterioration of filterability occurs at a temperature of 160° C [59] for T-5 fuel.

Increase of the fuel temperature leads to increase of its saturated vapor pressure. A fuel with low saturated vapor pressure, T-5 and T-6 for example, must be selected for reliable operation of the fuel system at high flight speeds. In order that T-5 fuel not boil at an altitude of 18,000 m, it is necessary that its temperature in the absence of excess pressure in the tank not exceed 110° C, and with an excess pressure of 30 KN/m^2 this temperature must not exceed 165° C (Figure 68). Considering the possible fuel temperature rise along the line from the EBP to the nozzles, the maximal permissible temperature of this fuel in the tanks must be 110 - 120° C.

We see from Figure 68 that the use of differential pressure in the fuel tank can increase the maximal permissible fuel temperature. But the magnitude of the differential pressure is limited by tank strength. Therefore the use of thermal insulation to reduce the heat influx into the fuel is more efficient.

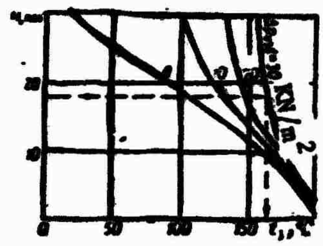


Figure 68. Temperature of boiling initiation for T-5 fuel as a function of flight altitude and differential pressure in tank

In the approximate <u>selection of the tank thermal insulation</u> we can neglect heating from external radiation, heat flux from the operating engines (and equipment), and radiation from the aircraft skin. This makes it possible to examine the heat transfer process accomplished only by convective heat exchange and thermal conduction.

The maximal amount of heat is transferred into the fuel in the course of the entire flight for the full tank case, for example, a standby tank group or a service tank. As the fuel level in the tank lowers an airspace is formed above the fuel, which serves as an insulating medium and reduces the heat transfer into the fuel. Since the forward, aft, and end walls of the tank also have an air insulation layer, we can neglect the heat transfer through these walls.

The process of heat transfer into the fuel (Figure 69) can be written in the form

$$Q = KS(t_{\text{в. с}} - t_{т}) = c_{т} E \rho_{т} \frac{\Delta t_{т}}{\Delta \tau}, \qquad (81)$$

where Q — is the heat flux, J/sec;

K — is the heat transfer coefficient, $J/m^2 \cdot sec \cdot deg$;

S — is the area of the upper and lower surfaces of the tank, m^2;

t_{bl} — is the air temperature in the boundary layer, °C;

t_f — is the fuel temperature in the tank, °C;

c_f — is the fuel specific heat, $J/kg \cdot deg$;

E — is the tank capacity, m^3;

ρ_f — is the fuel density, kg/m^3;

τ — is the time, sec.

If we consider only the heat transfer from the air to the aircraft skin and the thermal resistance of the insulation (neglecting the small thermal resistance of the wing skin and the tank wall), the heat transfer coefficient is expressed as

$$K=\frac{1}{\frac{1}{\alpha}+\frac{\delta}{\lambda}}\ J/m^2\cdot sec\cdot deg. \tag{82}$$

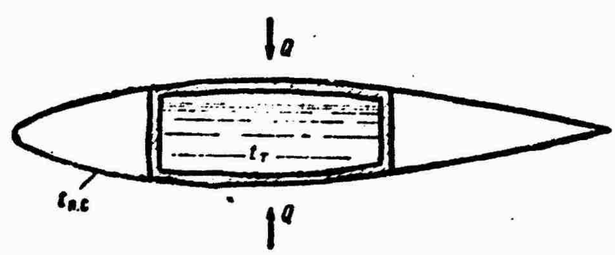


Figure 69. Diagram for calculating heat transfer into fuel in aerodynamic heating case

where α — is the coefficient of heat transfer from the air to the aircraft skin, $J/m^2 \cdot sec \cdot deg$;

δ — is the thermal insulation layer thickness, m;

λ — is the insulation thermal conductivity, $J/m \cdot sec \cdot deg$.

On the basis of (81) and (82) we obtain the equality

$$K = \frac{c_{\text{т}} E \rho_{\text{т}} \frac{\Delta t_{\text{т}}}{\Delta \tau}}{S(t_{\text{п.с}} - t_{\text{т}})} = \frac{1}{\frac{1}{\alpha} + \frac{\delta}{\lambda}}.$$

Hence the thermal insulation layer thickness is

$$\delta = \lambda \left[\frac{S(t_{\text{п.с}} - t_{\text{т}})}{c_{\text{т}} E \rho_{\text{т}} \frac{\Delta t_{\text{т}}}{\Delta \tau}} - \frac{1}{\alpha} \right] \text{м}. \tag{83}$$

The magnitude of the air temperature in the boundary layer with account for the Mach number, equal to the ratio of the flight speed to the speed of sound, is found from the formula

$$t_{\text{п.с}} = (t_H + 273)(1 + 0{,}2rM^2) - 273^\circ\text{C},$$

where t_H — is the ambient air temperature, °C;

r — is the recovery factor.

For the turbulent regime ($Re > 0.5 \cdot 10^6$) we assume that r = 0.89 and for the laminar regime r = 0.85.

The calculation is made for a constant temperature T_f = 45° C of the fuel in the tank. In actuality the heat influx is less because of the fuel temperature rise.

With fuel temperature variation from 45 to 90° C, its specific heat changes very little (see Appendix 6) and can be taken constant in the calculation.

The permissible value of $\frac{\Delta t_f}{\Delta \tau}$ depends on the flight duration τ and the maximal permissible fuel temperature t_f (Figure 70).

The value of the coefficient of heat transfer from the air to the wing skin is found from the formula which is familiar from the heat transfer course

$$\alpha = \frac{Nu\lambda}{l}\ J/m^2\cdot sec\cdot deg,$$

where Nu — is the Nusselt number;

l — is a characteristic dimension, in the present case the tank width, m.

Considering the upper and lower surfaces of the tank as flat plates, to determine the Nusselt number we use the criterial equation $Nu = 0.032\ Re^{0.8}$, in which the Reynolds number is defined by the formula

$$Re = \frac{vl}{\nu}.$$

The values of the thermal conductivities (which depend on the temperature) are given in Table 5 for temperatures from 50 to 100° C. In selecting a material for thermal insulation we must take into account the requirements imposed on this material: small value of the thermal conductivity, low density, sufficiently high mechanical strength, chemical stability, low hygroscopicity, ease of handling, high phase transformation temperature, and low cost. On the basis of the next-to-last requirement the direct use of certain materials with quite low value of the thermal conductivity (cork, cardboard) is not possible. Therefore it is advisable to use a multilayer insulation whose surface layer can withstand high temperatures, while the other layers are made from materials with lower values of the thermal conductivity.

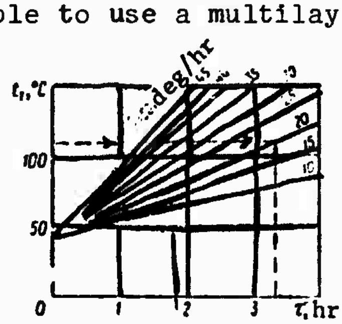


Figure 70. Determining permissible fuel heating rate as a function of its maximal permissible temperature and flight duration

The most acceptable materials for thermal insulation are the metaloceramic coatings, consisting of metallic oxides, carbides, nitrides, borides, silicides, and

TABLE 5. CHARACTERISTICS OF THERMAL INSULATION MATERIALS

Material	Thermal conductivity, λ, J/m·sec·deg	Density, ρ, kg/m^3
Cork sawdust	0.031	350
Glass wool	0.074	410
Slag wool	0.085	420
Sole leather	0.163	1,020
Linoleum	0.186	1,300
Asbestos	0.193	580
Rubber	0.232	920
Carbon steel	58.150	7.850
Aluminum	203.000	2.700

the refractory metals, and also certain types of glassplastics based on the phenolformaldehyde, silicone, and polyester resins.

The thickness of the insulation layer must not be too great; otherwise the structural weight increases.

The thermal conductivity of air is very low (at a pressure of 100 KN/m^2 and temperature 100° C, $\lambda = 305 \cdot 10^{-4}$ J/m · sec · deg). Therefore an air interlayer between the tank and the aircraft wing skin (if it can be provided) is the most effective and lightest-weight thermal protection.

The tank strength analysis is made like that of thinwalled vessels.

The required tank wall thickness δ is found from the formula

$$\delta = f\,\frac{3l}{2\sigma_{\text{в}}}\ \text{мм},$$

where f — is the safety factor (1.5 - 2.0);

Δp_t — is the maximal differential pressure on the tank wall, which depends on the pressurization pressure Δp_{ex} and the maximal inertial pressures Δp_i, N/m^2. The design value of Δp_t should not be less than 15 KN/m^2.

l — is the tank width or height (maximal value), mm;

σ_{ult} — is the ultimate strength of the material, N/m^2.

The inertial pressures are determined as the maximal permissible load factors for the given aircraft with the tank full. For civil aircraft the design coefficients will be n_y. Then for a tank with height h the inertial pressure on the bottom of the tank is $\Delta p_{i_y} = n_y h \gamma_f$. The values of p_{i_y} may amount to 40 - 60 KN/m^2.

Providing Fuel Supply to Engines at High Flight Speeds and Altitudes

Flights of supersonic airplanes at speeds twice that of sound and at altitudes above 15,000 - 16,000 meters complicate the operation of the fuel system to a considerable degree. As a result of fuel heating there is an increase of its saturated vapor pressure, which reduces the fuel system ceiling. Flights at high altitudes, with low atmospheric pressure, are also accompanied by significant fuel evaporation. At the high flight speeds the temperature of the unwetted tank walls may reach the fuel spontaneous ignition temperature, which reduces flight safety. The shift of the center of pressure of the aerodynamic forces during transition from subsonic to supersonic speeds leads to disruption of airplane trim and deterioration of the stability and controllability.

To improve the fuel system ceiling by reducing the saturated vapor pressure of the fuel, we must reduce the fuel temperature. For this purpose it is desirable to fuel the tanks of the supersonic airplane with cold fuel. Moreover, coolers (turbine or other types) may be used to cool the fuel in flight, and the use of fuel feed systems with series rather than parallel connection of the tank groups is possible since the long residence time of the fuel in the tank from which it is last used leads to increased fuel temperature rise.

The service tank should be located in the least heated part of the airplane, and the fuel should be used first from the tanks which are heated most. Realization of these recommendations is made difficult by the requirements for maintaining the airplane c.g. location.

The use of heavy fuel grades with lower saturated vapor pressure (types T-5 and T-6) and degassing of the fuel improve the operation of the fuel system at high flight speeds. Fuel degassing is performed with the aid of vapor-air separators (centrifugal or the settling type). The operation of the centrifugal vapor-air separator is based on the difference of the gas and liquid fuel specific weights when their mixture is imparted a rapid rotational motion. In this process the gases are easily separated from the liquid. The operation of the settling-type vapor-air separator is also based on the difference of the specific weights of the gases and the liquid fuel, but in this case the liquid is not given a rotational motion; rather the liquid settling principle is used. The simplest technique for separating gases from the fuel is to pump it along a closed circuit with a low pressure segment.

In addition to controlling the magnitude of the saturated vapor pressure of the fuel, steps can be taken to reduce the ABP cavitation pressure margin. To do this it is necessary to reduce the centrifugal pump impeller speed and output, and use pumps with good anticavitation properties.

Increase of the differential pressure in the tanks aids in increasing the fuel system ceiling. If the tanks are pressurized by inert gases, this also reduces evaporation losses and improves fire safety.

To ensure airplane stability and controllability during transition from subsonic to supersonic speeds and vice versa, use is made of trim fuel transfer, in which the center of gravity is shifted to follow the movement of the center of pressure. For this purpose the airplane must be equipped with a c.g. controller for determining the c.g. location in flight and a transfer controller, which obtains signals concerning the center-of-pressure location and the magnitude of the flight speed, which affects the center-of-pressure position. After comparing these signals the controller automatically transmits a command to activate the fuel tank boost pumps if necessary. After acceptable relationships between the center-of-pressure and center-of-gravity locations are reached, the pumps are deactivated.

In order to provide effective fuel transfer trimming, the tank groups must be located in the forward and aft sections of the fuselage, and special transfer manifolds with minimal hydraulic resistance and powerful boost pumps (high output) must be used.

Refueling Manifolds

Refueling Techniques

Refueling of the tanks may be accomplished by either gravity or pressure. In gravity type refueling the filtered fuel from the refueling unit is fed through a flexible hose and the delivery nozzle directly to the filler neck. The filler neck is located in the upper part of the tank, from which the fuel (in the case of grouping of the tanks) flows by gravity through connecting lines or flanges into the other tanks. The filler necks must permit the use of fueling nozzles of the dimensions specified by international standards. The time for filling all the tanks of the aircraft must not exceed 10 minutes.

Gravity fueling has the following disadvantages:

long fueling time owing to the low fuel flow velocity through the tank interconnecting lines and the additional time required to prepare for fueling (opening and closing the filler neck caps, dragging up the delivery hose, turning on and off the refueling unit pump);

positioning of servicing personnel at the filler necks, usually located on the upper surface of the wing. This requires the personnel to get up on the wing, move around, and get back down, which requires ladders, walkways, and long hoses. Movement of the servicing personnel and dragging of the hoses leads to damage of the wing finish. In the wintertime movement over the icy wing surface is hazardous for the personnel;

fuel evaporation and fire hazard;

entry of moisture and dirt into the tank.

The drawbacks of the gravity refueling system are eliminated when using pressure fueling. In this case the filtered fuel from the fueling unit is fed through a flexible hose to the tank fueling receptacles. This permits a high rate of fuel flow through the fueling manifold and the tank interconnect lines.

The flow rate during refueling must be at least 1500 liters/min through each refueling receptacle at a pressure not exceeding 4.5 kgf/cm^2. The refueling receptacles and delivery hose nozzles are made to dimensions specified by international standards.

Pressure fueling is accomplished through one or two fueling receptacles; therefore this type of fueling can be termed "centralized". To reduce the length of the delivery hoses, and for convenience in servicing, the fueling receptacles are located on the underside of the aircraft and this type of fueling has been termed "underwing fueling". In view of the high fuel flowrate, the considerable increase of the length of the entire refueling circuit, and the presence of several controlling devices, the hydraulic resistance of the refueling manifold is high. Therefore high pressure differentials must be supplied by the refueling unit pumps. This has led to the appearance of the term "pressure fueling".

Pressure fueling also has some disadvantages:

increase of the structural weight as a result of installation of the fueling system equipment in the airplane;

equipment complication (presence of controlling and protective devices);

impossibility of complete fuel servicing of the tank because of actuation of the limit level valves. Therefore arrangements for gravity fueling are also provided.

Pressure fueling can be accomplished using various schemes. One such scheme is shown in Figure 71. Fuel enters the refuel manifold through the fueling receptacle 1. Switches are activated to open the fueling valves 2. The valve positions are indicated by lights.

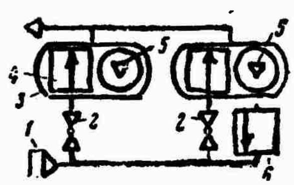


Figure 71. Pressure fueling manifold

After the tanks 3 are filled with fuel, the inductive level transmitters 5 send signals for automatic closure of the refueling valves. In case of inductive sensor malfunction there are float type level switches which back duplicate the commands for closure of the refueling valves. In case the valve fails, there are safety float-type level valves 4 which prevent overfilling of the tanks and fuel spillage through the vent lines. The vent valve 6 is provided for pumping the fuel from the hose. Indicators showing the fuel quantity in the tanks may be located on the refueling panel. Manual control of the refueling operation is provided in addition to automatic control.

The tank refueling order may be either concurrent or sequential. Sequential fueling of the tanks requires more time than does concurrent fueling. However, sequential fueling offers the possibility of filling only certain required tank groups rather than all the tank groups. The hydraulic resistance of the parallel manifold branches is lower in concurrent fueling than in sequential fueling. This makes it possible to use lower-power pumps in the fueling units. However, selective filling of any one group of tanks is impossible in this case. When using concurrent fueling of the tanks it is possible that they will become full at the same time or at different times.

System Components

The filler necks for gravity tank refueling consist of openings in the tank or in a refueling pipe, which are covered by caps. A screen may be installed inside the filler neck to prevent entry of foreign objects into the tank. In some cases the filler neck is surrounded by a scupper with a tube through which any fuel which is spilled during fueling is drained to the ground.

Many aircraft use filler necks with a quick-removable plug to reduce vehicle skin friction and for convenience in servicing.

Refueling receptacles of various types are used. The primary element of the receptacle is the valve, which is opened by the nozzle on the refueling unit delivery hose.

Level sensors provide a signal to close the fueling valve when the tank is filled with fuel. The sensor float rises and breaks the electrical circuit which holds the valve open.

Level valves are required to shut off fuel entry into the tank if the "shutoff" valves do not activate. They are installed in each tank group being refueled.

The refuel valves operate at high pressures. Therefore they are usually of the disk type. The disks may displace under the action of a screw driven by an electric motor or under the action of a lever mounted on the shaft of an electric actuator.

Analysis

In making the design calculation of the refueling manifold, we start from the given refueling duration and the known volume of the tanks being filled. After selecting the refueling manifold layout, we can determine the line lengths and magnitudes of the local resistance coefficients. We then need to find the line diameters for known characteristics of the refueling unit pump or select the pump for given values of the refueling manifold line diameter.

The check calculation is made for a specific aircraft type. In this case the line diameters and other geometric characteristics of the refueling manifold must be known. The unknowns are the refueling time and the magnitude of the pressure differential Δp_p which the refueling pump must develop in this case.

Figure 72 shows the computational scheme for sequential fueling. The number of tanks (tank groups) may vary. The line lengths l, local resistance coefficients ζ, and tank volumes E are known. Let us assume that the fuel first enters tank 1, fills it, and then enters tank 2.

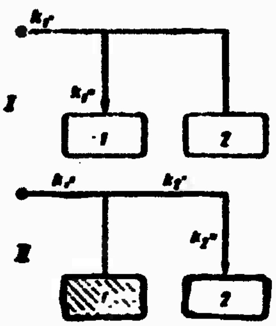


Figure 72. Sequential fueling scheme

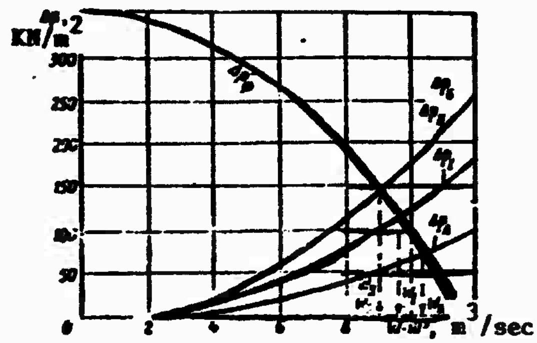


Figure 73. Determining theoretical flow rates for sequential fueling

We shall describe the technique for the check calculation. The total duration of the sequential tank fueling will be

$$\tau=\tau_1+\tau_2+\ldots+\tau_i,$$

where τ_1, τ_2, ..., τ_i — are the fueling durations of the corresponding tanks.

The tank fueling duration is

$$\tau_i=\frac{E_i}{W_i},$$

and the total fueling duration is

$$\tau=\frac{E_1}{W_I}+\frac{E_2}{W_{II}}+\ldots+\frac{E_i}{W_i}. \quad (84)$$

Since the volumes E_1, E_2, ..., E_i of the tanks being filled are known, to determine the fueling duration we need to have the values of the fuel flowrates W_1, W_{II}, ..., W_i. To find these values we must calculate the hydraulic resistances of the refueling manifolds using (24) and plot in Figure 73 their dependence on the fuel flowrate. The values of the specific weight are determined for the lowest fuel temperature. Considering the thermal insulation of the storage tanks and the temperature rise of the fuel during pumping, the design fuel temperature may be taken equal to minus 50° C.

We also plot on the figure the refueling unit pump head curve with account for the resistance of the delivery manifold in the segment from the pump to the fueling receptacle, and the pressure required to supply fuel from the pump to the fueling receptacle.

The projections on the abscissa axis of the points of intersection of the curves indicate the values of W_I, W_{II}, ..., W_i. Then we can calculate the durations τ_1, τ_2, ..., τ_i and the total fueling duration τ.

In the design calculation we specify the fueling duration τ_i for a tank of capacity E_i and find the flowrate

$$W_i=\frac{E_i}{\tau_i}.$$

After this the determination of the fueling manifold line diameter and the pump selection are made similarly to the procedure described in determining the engine feed manifold line diameter and selecting the airframe boost pumps.

In the case of concurrent refueling with nonsimultaneous filling of the tanks, as certain tanks are filled the refuel valves block the entry of fuel, while topping of the other tanks will continue. Initially, when fuel is entering all the tanks, the computational scheme corresponds to version A in Figure 74. Then fuel starts to flow into the tanks in accordance with version B. Let us examine the check calculation procedure, assuming that tank 1 is filled first and then tank 2.

The total fueling time is

$$\tau = \tau_A + \Delta\tau_B. \qquad (85)$$

where τ_A — is the fueling time for tank 1 with fuel feed in accordance with version A

$$\tau_A = \frac{E_1}{W_{A1''}};$$

$\Delta\tau_B$ — is the time for topping tank 2 with fuel feed in accordance with version B

$$\Delta\tau_B = \frac{\Delta E_2}{W_{B2''}} = \frac{E_2 - \tau_A W_{A2''}}{W_{B2''}}.$$

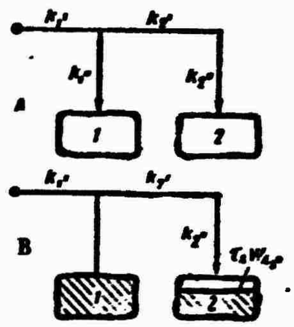


Figure 74. Concurrent fueling scheme with nonsimultaneous filling of tanks

To solve the problem we must find the flowrates $W_{A1''}$, $W_{A2''}$, $W_{B2''}$ using the following procedure. Drawing contours on the computational diagram (Figure 75), we find for fueling version A the reduced hydraulic resistance coefficient of the fueling manifold, using Formula (35) for the mixed line connection

$$k_A = k_{[illegible]} = \sqrt{k_{1'}^2 + \frac{1}{\frac{1}{k_{1''}} + \frac{1}{\sqrt{k_{2'}^2 + k_{2''}^2}}}}.$$

The notations for the contours and elements correspond to their reduced hydraulic resistance coefficients.

If refueling of all tanks is accomplished through a single receptacle, then we must examine the corresponding configuration (Figure 76) and find

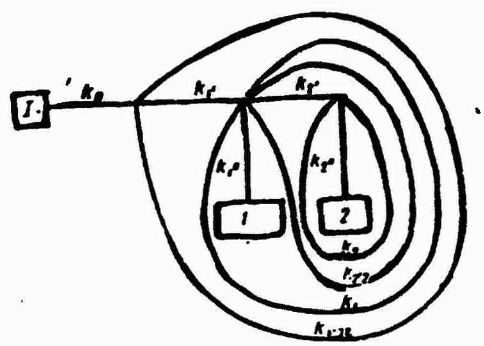


Figure 75. Determining reduced hydraulic resistance coefficient of complex manifold:

I — tanker

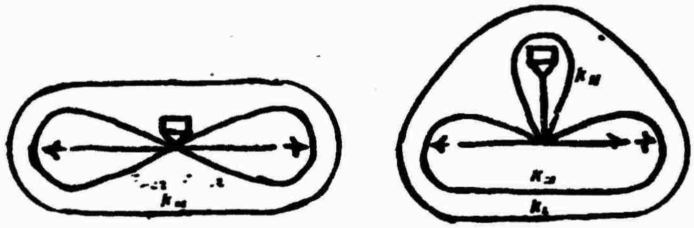


Figure 76. Determining reduced hydraulic resistance coefficient of complex manifold when aircraft is fueled through a single receptacle

the value of the reduced hydraulic resistance coefficient

$$k_A = k_L = \sqrt{k_N^2 + \left(\frac{k_{\text{мат}}}{2}\right)^2}.$$

Taking various values of the flowrate W, we use (24) to find the manifold hydraulic resistance Δp_A (see Figure 75). The intersection of the refueling manifold hydraulic resistance curve with the refueling unit pump head characteristic (referred to the point where the refueling receptacle is located) makes it possible to find the fuel flowrate $W_A = W_{A1'}$ and the pump pressure rise Δp_p.

For parallel connection of the element $k_{1''}$ and the contour $k_{2'2}$ we have the condition $W_{A1'} = W_{A1''} + W_{A2'2}$.

On the basis of (33) and (34) we can find the flowrates

$$W_{A1''} = \frac{k_{2'2}}{k_{2'2} + k_{1''}} W_{A1'};$$

$$W_{A2''} = W_{A2'2} = \frac{k_{1''}}{k_{2'2} + k_{1''}} W_{A1'}.$$

Examining refueling configuration B, we establish the relation for the reduced hydraulic resistance coefficient $k_B^2 = k_{1'}^2 + k_{2'2}$

and find the hydraulic resistances Δp_B. We obtain the value of the flowrate $W_B = W_{B2'} = W_{B2''}$ graphically (see Figure 73).

In the design calculation we know the total refueling time and the refueling time for each refueling version. Knowing the tank capacities, we can calculate the required flowrates. The line diameters can be found by the method described for finding the diameter of the engine feed line and selection of the boost pump.

<u>Concurrent refueling with simultaneous filling of the tanks is</u> a particular case of the problem examined above. Now $\Delta\tau_B = 0$ and the time for filling all tanks and each tank individually is

$$\tau=\tau_A=\tau_1=\tau_2=\frac{\Sigma E}{W_A}=\frac{E_1+E_2}{W_A}. \qquad (86)$$

Let us find the condition under which simultaneous filling of the tanks is possible. The fuel flowrate for each manifold branch leading to a tank will be $W_{1''}$, $W_{2''}$. Since

$$\tau_1=\frac{E_1}{W_{1''}} \text{ и } \tau_2=\frac{E_2}{W_{2''}}, \text{ then } \frac{E_1}{W_{1''}}=\frac{E_2}{W_{2''}}=\frac{\Sigma E}{W_A}$$

or

$$W_{1''}=\frac{E_1}{\Sigma E}W_A; \qquad (87)$$

$$W_{2''}=\frac{E_2}{\Sigma E}W_A. \qquad (88)$$

Consequently, for similar filling the flowrates must be proportional to the tank capacities. Then

$$W_{2''}=\frac{\Sigma E-E_1}{\Sigma E}W_A=\left(1-\frac{E_1}{\Sigma E}\right)W_A. \qquad (89)$$

Since for a parallel connection the product of the reduced hydraulic resistance coefficient by the flowrate is a constant quantity [see (31)], we can write $k_{1''}W_{1''} = k_{2'2}W_{2'}$.

Using (87) and (89), we obtain

$$k_{1\cdot}\frac{E_1}{\sum E}W_A = k_{2\cdot 2}\frac{\sum E - E_1}{\sum E}W_A.$$

Then

$$k_{1\cdot}E_1 = k_{2\cdot 2}(\sum E - E_1). \qquad (90)$$

Consequently, in the case of simultaneous filling for parallel connections the condition that the product of the reduced hydraulic resistance coefficients by the tank volume being filled be constant must be satisfied, i.e.,

$$kE = \text{const.} \qquad (91)$$

In the design calculation we must start from the condition (91). Taking several values of the line diameters, we can calculate the corresponding values of the reduced hydraulic resistance coefficients of all the elements of the refueling manifold and compare them with the required values, which are found from (90)

$$\frac{k_{1\cdot}}{k_{2\cdot 2}} = \frac{\sum E - E_1}{E_1} = \frac{\sum E}{E_1} - 1.$$

The technique for selecting the refueling unit pump is the same as that described in examining sequential refueling.

Drain Manifolds

Drainage Techniques

Fuel is drained with the airplane on the ground through the tank and manifold drain cocks in order to perform major maintenance and certain inspection operations. Fuel dumping in flight is provided on certain types of aircraft when their landing weight is greater than the permissible value. Dumping may also be used in emergency cases to change the center of gravity.

Fuel drainage can be accomplished by gravity, using pressurization, and with the aid of pumps. The use of a particular technique depends on the configuration of the aircraft and the location of tanks and engines. The fuel in tanks located far from the engines is drained by gravity or pressurization. In order to reduce the fire hazard, the fuel must be pumped from tanks located near operating engines to a safe dumping location, for example at the wingtips.

Figure 77 shows a typical fuel draining arrangement. The fuel drains from tank 5 through the valve 6 by gravity. From tank 1 the fuel is pumped to the wingtip by the pump 2. During draining, the dump manifold valve 4 is closed.

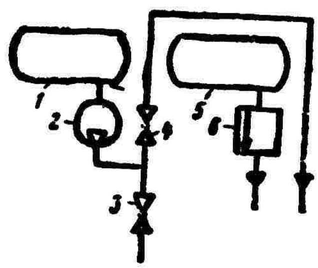


Figure 77. Fuel dumping manifold

Analysis

In the check calculation we determine the time for draining the fuel from tanks of known capacity E for given dimensions of the drain manifold lines and known pump characteristics

$$\tau = \frac{E}{W_{cp}}, \quad (92)$$

where W_{av} — is the average fuel flowrate, which must be no less than 1500 liters/min for tank capacities up to 10,000 liters; 1700 liters/min for tank capacities to 20,000 liters; and 2000 liters/min for tank capacities greater than 20,000 liters.

Let us examine the calculation of fuel dumping in flight by gravity from a tank of arbitrary shape (Figure 78).

The fuel flowrate from the tank $W = \frac{dE}{d\tau}$, then

$$d\tau = \frac{dE}{W}. \qquad (93)$$

The volume of a fuel element in the tank can be expressed through its area F and height dy

$$dE = -F\,dy. \qquad (94)$$

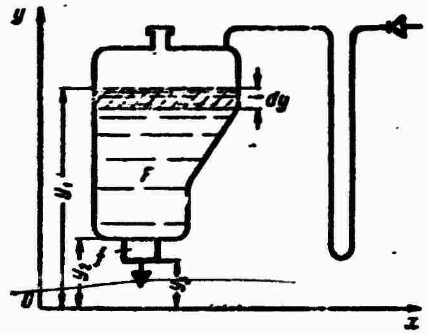


Figure 78. Gravity fuel dumping

The minus sign is taken since dy is a negative quantity during fuel dumping for the selected direction of the y axis.

Substituting (94) into (93), we obtain

$$d\tau = \frac{dE}{W} = -\frac{F\,dy}{W},$$

hence

$$\tau = \int_{y_1}^{y_2} -\frac{F\,dy}{W}.$$

For a tank in the form of a parallelepiped F = const. For a tank of arbitrary shape we can determine the average area

$$F_{cp} = \frac{E}{y_1 - y_2}.$$

Then, after changing the limits of integration

$$\tau = F_{cp} \int_{y_2}^{y_1} \frac{dy}{W}.$$

The fuel flowrate through the drain line is

$$W = fV,$$

where f — is the area of the drain line;

V — is the draining velocity.

This velocity is found from the formula

$$V = \varphi\sqrt{2gy},$$

where ϕ — is the velocity coefficient; for short lines $\phi = 0.82$.

As a result we obtain

$$\tau = F_{cp}\int_{y_s}^{y_1}\frac{dy}{fV} = \frac{F_{cp}}{f\varphi\sqrt{2g}}\int_{y_s}^{y_1}\frac{dy}{\sqrt{y}}.$$

Calculation of the draining time reduces to solving the problem of discharge from a tank with variable head. Performing the integration, we find

$$\tau = \frac{2}{\varphi\sqrt{2g}}\cdot\frac{F_{cp}}{f}(\sqrt{y_1}-\sqrt{y_2}).$$

After substituting the values $\phi = 0.82$ and $g = 9.81\ m/sec^2$, we obtain

$$\tau = 0{,}55\frac{F_{cp}}{f}(\sqrt{y_1}-\sqrt{y_2})\ \text{sec.} \qquad (95)$$

If fuel draining is accomplished by pressurizing at the differential pressure Δp_{diff}, then the energy height of the liquid column is increased by the magnitude $\Delta p_{diff}/\gamma_f$. Then

$$\tau = 0{,}55\frac{F_{cp}}{f}\left(\sqrt{y_1+\frac{\Delta p_{изб}}{\gamma_т}}-\sqrt{y_2+\frac{\Delta p_{изб}}{\gamma_т}}\right)\ \text{sec.} \qquad (96)$$

When gravity draining fuel from a tank with a short line we can assume that $y_2 = y_3 = 0$. Then

$$\tau = 0{,}55\frac{F_{cp}}{f}\sqrt{y_1}\ \text{sec}.$$

Replacing

$$y_1 = \frac{E}{F_{cp}},$$

we obtain the draining time

$$\tau = 0{,}55\frac{F_{cp}}{f}\sqrt{\frac{E}{F_{cp}}} = \frac{0{,}55}{f}\sqrt{EF_{cp}}\ \text{sec}. \qquad (97)$$

In the design calculation we determine the dimensions of the drain line for a given draining time.

For gravity draining

$$f = \frac{0{,}55}{\tau}\sqrt{EF_{cp}}\ м^2,$$

and for pressurized draining

$$f = \frac{0{,}55}{\tau}\left(\sqrt{y_1 + \frac{\Delta p_{изб}}{\gamma_T}} - \sqrt{y_2 + \frac{\Delta p_{изб}}{\gamma_T}}\right) м^2.$$

When pumps are used for fuel draining (the ABP are usually used for this purpose), in the check calculation we wish to determine the draining time for given values of the drain manifold diameters and known boost pump characteristic.

Taking various values of the flowrates W_{av}, we calculate the Reynolds number, the frictional resistance coefficient λ, the equivalent ζ_{eq} and reduced k hydraulic resistance coefficients, and finally the hydraulic resistance Δp_{man}.

The intersection of the hydraulic resistance curve with the boost pump head curve (Figure 79) indicates the operating point and the corresponding value of the draining flowrate W_{av}.

The draining time is found from (92) as a function of the value of W_{av}.

In the check calculation we start from the given draining time and find the values of the drain manifold diameters (assuming that the boost pump is already selected and its head curve is known). The required flowrate is found from the formula $W_{av} = \frac{E}{\tau}$. This value is laid off on the boost pump head curve. The remainder of the solution is analogous to the solution of the problem of finding the engine feed line diameters.

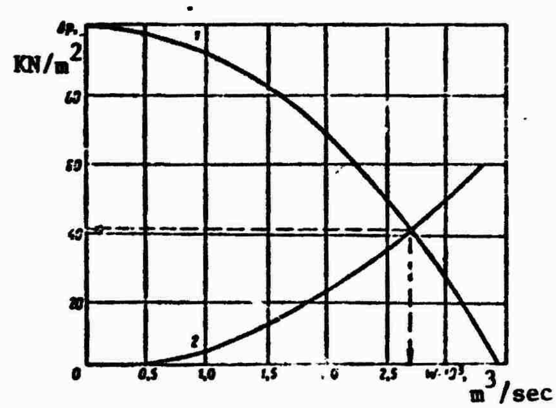


Figure 79. Determining fuel flowrate when dumping using ABP: 1 — ABP head curve; 2 — hydraulic resistance of dump line

Fuel Tank Venting Systems

Configurations

The purpose of the fuel tank vent system is to maintain the pressure in the tank air space within limits which will provide reliable fuel feed to the engines and permit refueling and fuel dumping. As fuel is used from a tank it is necessary that the tank be filled with air or inert gases. If this is not done it becomes difficult for the fuel to flow from the tank and tank collapse may occur. During refueling of the tanks, particularly pressure refueling, it is necessary to provide the possibility for the air to leave the tank. Otherwise air blocks will form and deformation of the tank is possible.

The <u>open fuel tank vent system</u> is designed so that the tanks are connected to the atmosphere through external vent line fittings (see Figures 64 and 66). In some cases the vent fittings are located in a high-pressure region or are made to face into the airstream. This arrangement provides pressurization of the tank and creates a pressure higher than atmospheric in the tank air space.

On aircraft with a large number of fuel tanks, venting is accomplished by means of individual or collective connection of the tanks to the atmosphere (Figure 80). When using individual connection of the tanks with the atmosphere, the weight of the vent system lines is less but the pressure in the tank air space may be different in the different tanks. When collective manifolding of the tanks to the atmosphere is used, the line weight will be somewhat less than when using individual venting but the pressure in the tanks is more uniform.

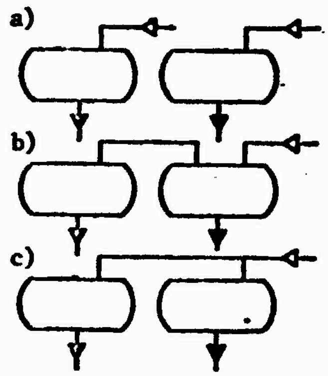


Figure 80. Fuel tank venting: a — individual; b, c — manifolded venting with series and parallel connections

In order to provide venting of a completely filled fuel tank and prevent loss of fuel, the vent line is attached to the tank at the highest point. The external fitting of this line is usually located near the upper part of the aircraft so that dirt will not get into the fuel system along with air when the aircraft is on the ground.

Looped segments are used in the lines to prevent loss of fuel through the vent lines during aircraft maneuvering. However, fuel collects in the lower parts of these looped segments. In order to recover the fuel, small vent tanks or tank compartments are sometimes installed, and when these are overfilled the fuel is automatically pumped back into the tanks.

On some types of aircraft which have considerable wing dihedral the vent lines are duplicated to provide better venting reliability. One (primary) vent line is connected to the highest point of the tank; the other (secondary) line is connected to another part of the tank and somewhat lower. If during aircraft maneuvers the higher point of the completely filled tank is lowered and venting through the primary line becomes ineffective, the secondary line provides venting. The installation of a check valve in the looped secondary vent line prevents fuel loss when the tank is completely full.

When necessary, a minimal pressure in the fuel tank vent system is provided by installing vacuum-relief valves, which open and permit air to enter the system when a slight suction develops (0.02 - 0.05 kgf/cm^2). These valves are located inside the wing. If the external fitting freezes over, these valves open and provide venting.

The airstream ram pressure depends on the air mass density ρ_a and the flight velocity V

$$q=\frac{\rho_a V^2}{2}\ \mathrm{N/m^2}.$$

At low altitudes even at subsonic flight speeds the magnitude of the ram pressure reaches 50 - 60 KN/m^2, which may be excessive. To limit the magnitude of the ram pressure, relief valves are installed in the venting system or the degree of utilization of the ram pressure is altered. The magnitude of the air ram pressure used to increase the pressure in the tank air space is

$$p_q=k_\phi q,$$

where k_ϕ — is the ram pressure utilization coefficient, which depends on the bevel angle ϕ of the end of the vent line external fitting: $k_\phi \approx \cos 1.5\,(90 - \phi^\circ)$.

The maximal ram pressure recovery will occur for the fitting which faces directly into the stream (Figure 81a). If the fitting is beveled (Figure 81b), the ram pressure recovery will decrease.

At subsonic flight speeds the air pressure downstream of the vent line external fitting is

$$p_{в}=p_{H}+p_{q}=p_{H}+k_{q}q.$$

At supersonic speeds a normal shock forms ahead of the vent line external fitting, and as a result the differential pressure in the tank air space may exceed the allowable value. In order to reduce

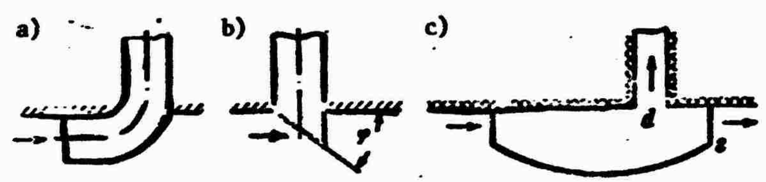


Figure 81. Vent line external fittings:
a, b — for subsonic aircraft; c — for supersonic aircraft

this pressure, use is made of relief valves or through-flow external vent line fittings (Figure 81c). As a result of air flow through section e the pressure at supersonic speeds is reduced, while at subsonic speeds the pressure will be adequate as a result of throttling of the velocity at the diffuser section d.

The <u>closed fuel tank venting system</u> is provided by pressurizing the tanks with air or gas. The tank air space is not open to the atmosphere. The excess pressure may be created by supplying air from the engine compressor, air or inert gases from on-board bottles, or from a special system.

Although the closed venting systems make it possible to maintain the required pressure in the tank air space, the complexity of the equipment of these systems, the considerable weight of the components, and poor self-sufficiency (in the case of on-board pressure sources in the form of bottles) are disadvantages, and therefore such systems have not been widely used on civil aircraft flying at altitudes up to 13,000 m. It is possible that such systems will find application in providing fuel tank venting at higher altitudes, where the atmospheric air density is very low.

The combined venting system differs from the open system in that differential pressure in the tank may be obtained by pressurizing the tanks by air bled from the engine compressor (Figure 82). An assembly of relief and check valves provides automatic switchover of system operation to that pressurization source which creates the required pressure.

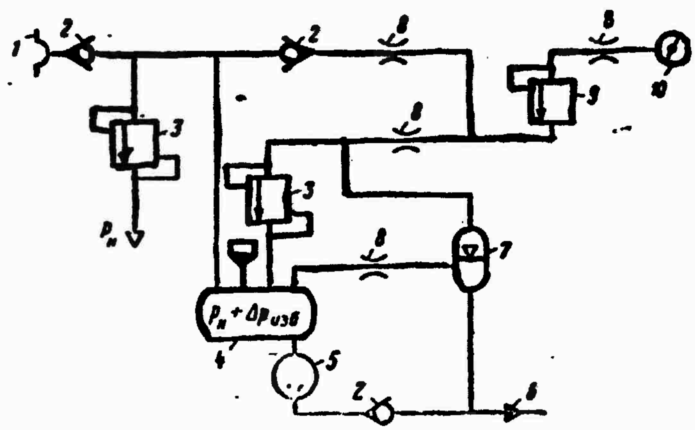


Figure 82. Fuel accumulator pressurization and combined vent system:

1 — air intake from atmosphere; 2 — check valve; 3 — pressurization safety valves for tank and fuel accumulator; 4 — tank; 5 — ABP; 6 — fuel feed line to engine; 7 — fuel accumulator; 8 — restrictor; 9 — reducing valve; 10 — compressed air intake from engine

Two types of relief valves are encountered: those which provide a constant differential pressure at all altitudes and those which provide a constant pressure at flight altitudes near the design altitude (Figure 83). Air/air radiators are sometimes installed to cool the air coming from the compressor.

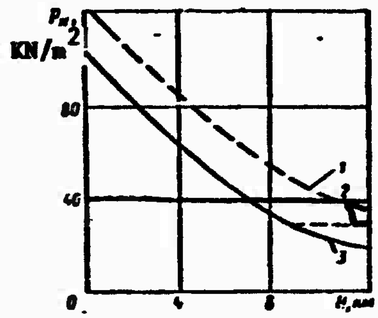


Figure 83. Pressure variation in tanks:

1 — with constant pressure differential at all altitudes; 2 — with constant pressure at flight altitudes near the design altitude; 3 — air pressure in standard atmosphere

Analysis

Open System

The calculation of the open fuel tank venting system reduces to determining the vent line external fitting bevel angle and the line diameter.

The vent line external fitting bevel angle can be found from the formula

$$\varphi = 90 - \frac{\arccos k_\varphi}{1,5} = 90 - \frac{\arccos \frac{p_q}{q}}{1,5} \text{ deg.} \qquad (98)$$

In order that the pressure in the tank air space not be excessive, the value of the ram pressure q is taken for the limiting permissible flight speed $V_{max\ max}$. For airplanes with gas turbine engines, which fly at high speeds, the design values of q will be at sea level at the speed V_{maxmax}. For airplanes flying at supersonic speeds we must determine the maximal value of q for several altitudes with account for the speed limitations based on Mach number and temperature.

The design cases for determining the vent line diameter are: fuel dumping from the tank, pressure refueling, airplane emergency descent.

Fuel usage from the tank when fuel is fed to the engines takes place with comparatively low flowrates, and the calculated vent line diameter will be small. Therefore the calculation should be made for the fuel dumping case for those airplanes which are equipped for dumping fuel in flight (the flowrate in this case is far greater than the flowrate to the engines).

Let us examine the tank which is most distant from the vent line external fitting (Figure 84). Examining the sections P-P and T-T (probe and tank), we can write the Bernoulli equation

$$p_H + p_q + y_П \gamma_в = p_H + \Delta p_{изб} + y_Б \gamma_в + \frac{\gamma_в V_Б^2}{2g} + \Delta p_{г.в} + \Delta p_{в.в}. \qquad (99)$$

Here the subscript a applies to the air.

Neglecting the quantities

$$\frac{\gamma_в V_Б^2}{2g}, \quad \Delta p_{в.в}, \quad (y_Б - y_П)\gamma_в$$

and solving (99) for the pressure p_q, we obtain

$$p_q = \Delta p_{изб} + \Delta p_{г.в},$$

i.e., the ram pressure required to overcome the hydraulic resistance and create the differential pressure in the tank.

Substituting in place of Δp_{ha} its value from (24) and taking $W_a = W_f = W$, we find

$$p_q = \Delta p_{изб} + c^2 k^2 W^2 \gamma_в.$$

hence

$$k^2 = \frac{p_q - \Delta p_{изб}}{c^2 W^2 \gamma_в} = \frac{\zeta_э}{d^4}.$$

Then

$$d = \sqrt[4]{\frac{\zeta_э c^2 W^2 \gamma_в}{p_q - \Delta p_{изб}}}. \tag{100}$$

If the vent line is a complex line with variable air velocity, the equivalent hydraulic resistance coefficient should be written in the form

$$\zeta_э = \sum \zeta_i \left(\frac{W_i}{W}\right)^2 \cdot \left(\frac{d}{d_i}\right)^4. \tag{101}$$

Since in order to determine the vent line diameter we need to know the values of the equivalent hydraulic resistance coefficient, which depend on the unknown line diameter, the problem can be solved by successive approximation.

It is recommended that a series of diameters d_{ass} be assumed in order to simplify the calculation. After determining from (100) for each value of d_{ass} the available diameter d_{avail}, we plot the curves of $d_{avail} = f(d_{ass})$ (Figure 85. Drawing the straight line $d_{avail} = d_{ass}$, at the points where it crosses the curves $d_{avail} = f(d_{ass})$ we

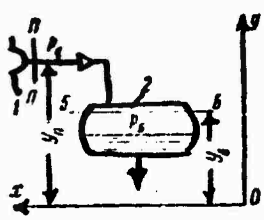


Figure 84. Diagram for analysis of vent line during fuel usage from tank:

1 — air intake from atmosphere; 2 — tank

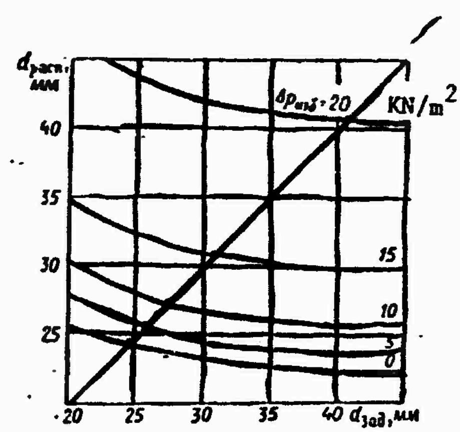


Figure 85. Graphoanalytical method for determining vent line diameter when using fuel from tank

find the values of the vent line diameter.

Figure 86 shows typical values of the vent line diameters for dumping fuel in flight as a function of the differential pressure (curve 1). To obtain large differential pressures it is necessary to have large vent line diameters. On the basis of the resulting differential pressure, we obtain the required vent line diameter.

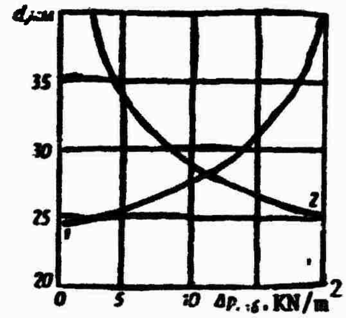


Figure 86. Determining vent line diameter:

1 — when using fuel from tank; 2 — when pressure refueling

During pressure fueling the fuel enters tank 1 (Figure 87) under pressure, and the air escapes through the vent line to the atmosphere. Writing the Bernoulli equation for sections T-T and P-P, we obtain

$$p_H + \Delta p_{изб} + y_Б \gamma_в + \frac{\gamma_в V_Б^2}{2g} =$$

$$= p_H + y_П \gamma_в + \frac{\gamma_в V_П^2}{2g} + \Delta p_{г.в} + \Delta p_{в.в}. \qquad (102)$$

Neglecting the quantities $\frac{\gamma_a V_T^2}{2g}$ and Δp_{ca}, and solving (102) for the differential pressure in the tank, we find

$$\Delta p_{изб} = \Delta p_{г.в} + \frac{\gamma_в V_П^2}{2g} - (y_Б - y_П)\gamma_в.$$

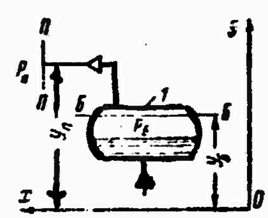


Figure 87. Diagram for analysis of vent line during pressure refueling

Neglecting the quantity $(y_T - y_P)\,\gamma_a$ and replacing

$$\Delta p_{\text{г.в}} + \frac{\gamma_{в} V_{п}^2}{2g} = \frac{c^2(\zeta_0 + 1) W^2 \gamma_{в}}{d^4},$$

we obtain

$$\Delta p_{\text{изб}} = \frac{c^2(\zeta_0 + 1) W^2 \gamma_{в}}{d^4},$$

hence

$$d = \sqrt[4]{\frac{c^2(\zeta_0 + 1) W^2 \gamma_{в}}{\Delta p_{\text{изб}}}}. \tag{103}$$

Formulas (100) and (103) for finding the vent line diameter have the same form; therefore the solution of this problem is analogous to that of the preceding problem (Figure 88).

Figure 86 shows the typical variation of vent line diameter for pressure refueling as a function of the differential pressure in the tank (curve 2). Large vent line diameters must be used to prevent high differential pressures in the tank.

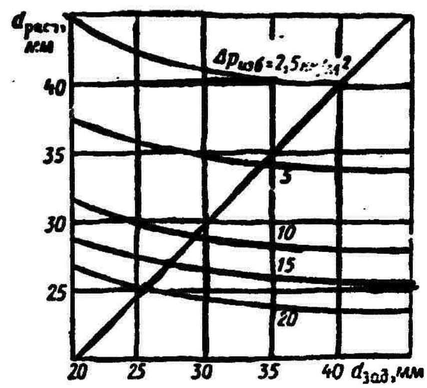


Figure 88. Graphoanalytical method for determining vent line diameter for pressure refueling

During airplane emergency descent the atmospheric pressure increases rapidly, and an adequate amount of air must be let into the tanks so that they will not collapse as a result of the difference between the pressures outside and inside the tanks.

In this case the vent line diameter can be found from the formula suggested by Polikovskiy [42]

$$d = 0{,}01\sqrt{E_б \frac{V_y}{V}} \cdot \sqrt[4]{\frac{\zeta_э}{k_э}}\ м,$$

where E_t — is the total tank capacity, m^3;

V_y — is the emergency descent vertical speed;

V — is the horizontal flight speed.

For civil aircraft the emergency descent vertical speed is comparatively low and the calculated required vent line diameters obtained in this case are not always the governing values. For acrobatic aircraft which permit steep, even vertical dives, this case is usually governing.

Combined System

Analysis of the combined vent system (see Figure 82) reduces to determining the relief valve pressure setting, selecting the reducing valves and restrictors, finding the line diameters and the maximal required air mass flowrate from the engine to pressurize the tanks and the fuel accumulator.

The service tank pressurization relief valve pressure setting Δp_{tv} is determined by the magnitude of the permissible differential pressure in the tank: $\Delta p_{tv} = \Delta p_{diff\ per}$.

The fuel accumulator pressurization relief valve setting Δp_{av} must be calibrated to the pressure

$$\Delta p_{к.в} = \Delta p_{в.з} + \Delta p_{г.к} > \Delta p_{к.б},$$

where Δp_{aa} — is the pressure in the accumulator air chamber;

Δp_{hv} -- is the hydraulic resistance in the segment from the valve to the accumulator air chamber.

In case of ABP failure the fuel accumulator must provide an air pressure Δp_{aa} on the membrane equal to the required fuel pressure.

On the basis of (53), we have

$$\Delta p_{\text{н.а}} = p_{t\,4/1} + \Delta p_{\text{нагнет.н-на}} - (y_{\text{а}} - y_{\text{Д}})\gamma_{\text{т}} + \Delta p_{\text{н.т}} + \\ + c^2 k^2_{\text{ман}} \gamma_{\text{т}} (W_{\text{да}} + W_{\text{пер}})^2 \text{ н/м}^2,$$

where $y_a - y_E$ — is the accumulator height above the EBP, which can be taken equal to $y_P - y_E$;

k_{man} — is the reduced hydraulic resistance coefficient of the engine feed manifold in the segment from the accumulator to the EBP.

Knowing the value of the air pressure downstream of a given engine compressor stage and the relief valve pressure setting, we select the reducing valves and restrictors which provide the required pressurization differential pressure. Knowing the pressure losses in the pressurizing manifold segments, we find the line diameters from the relation

$$\Delta p_{ri} = c^2 \frac{\zeta_{ai} \gamma_{\text{в}} W^2}{d_i^4},$$

hence

$$d_i = \sqrt[4]{\frac{c^2 \zeta_{ai} \gamma_{\text{в}} W^2}{\Delta p_{ri}}}.$$

The required air flowrates are as follows.

The volumetric air flowrate required to expel the fuel from the accumulator is

$$W_{\text{a}} = W_{\text{да}} + W_{\text{пер}} = W'_{\text{д}} \text{ м}^3/\text{sec}.$$

The mass flowrate is

$$G_{\text{a}} = W'_{\text{д}} \gamma_{\text{в}} \text{ N/sec}.$$

The specific weight of the air is

$$\gamma_{\text{в}} = \frac{p_{\text{в.а}}}{R_{\text{в}} T_{\text{в}}} \ \text{N/м}^3,$$

where R_a — is the gas constant of air, J/kg · deg;

T_a — is the air temperature, °K.

The air pressure p_{aa} is found from the relation

$$p_{\text{в.а}} = p_H + \Delta p_{\text{в.а}}.$$

The volumetric flowrate when the air is bled into the tank through the accumulator pressurization relief valve is

$$W_{\text{стр}} = \frac{1}{ck_e} \sqrt{\frac{\Delta p_{\text{к.а}} - \Delta p_{\text{к.б}}}{\gamma_{\text{в}}}},$$

where k_e — is the reduced hydraulic resistance coefficient of the line segment from the accumulator pressurization relief valve to the tank.

The mass flowrate when the air is bled into the tank through the accumulator pressurization relief valve is

$$G_{\text{стр}} = W_{\text{стр}} \gamma_{\text{в}} = \frac{1}{ck_e} \sqrt{(\Delta p_{\text{к.а}} - \Delta p_{\text{к.б}}) \gamma_{\text{в}}}.$$

The volumetric air flowrate for pressurizing the tank is equal to the fuel flowrate $W_t = W_{eng}$. Then the mass flowrate is $G_t = W_{eng}\gamma_a$. The maximal required engine air flowrate for pressurizing the tanks is

$$G_{\text{над}} = G_{\text{а}} + G_{\text{стр}} + G_{\text{б}} = (W'_{\text{д}} + W_{\text{дч}}) + \frac{1}{ck_e} \sqrt{(\Delta p_{\text{к.а}} - \Delta p_{\text{к.б}}) \gamma_{\text{в}}}.$$

Fuel Control System

Control of the fuel system involves activating the equipment which creates the fuel flow and organizing the direction of this flow along the required path.

Organization of the fuel flow direction is accomplished by cocks and valves. The difference between these devices is that cocks require controls while valves operate automatically when the definite conditions for which they are set are reached.

The cocks and valves are required to provide sealing in the closed position, fast actuation, and low hydraulic resistance in the open position.

With regard to purpose, cocks are divided into shutoff and selector types. With regard to construction, they come in a large number of varieties. They may be of the plug, sleeve, disk, plunger, and other types (Figure 89).

With regard to purpose, the valves are divided into check, relief, reducing, unloading, vent, disconnect, and so on. Valves actuate after disruption of the balance between the force acting on

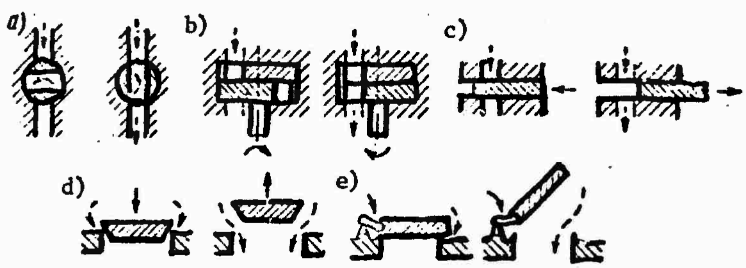


Figure 89. Types of cocks:
a — plug; b — slide; c — disk; d, e — poppet

the valve from the liquid or gas and the force acting from the spring, aneroid capsule, valve weight (ball, plate), moment of pendulous weight, float lift force, and so on (Figure 90).

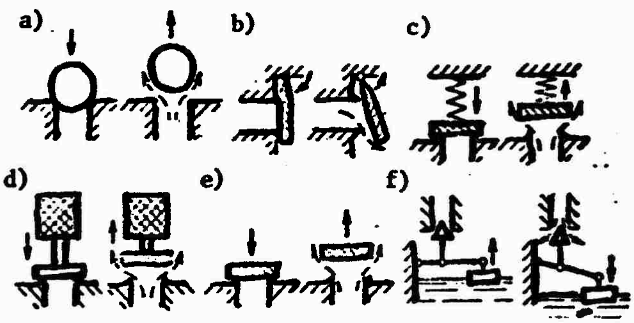


Figure 90 : Types of valves:
a — ball; b — pendulous; c — spring; d — electromagnetic ; e — poppet; f — float

Control of the fuel system components may be direct or remote.

Direct control is used for opening and closing cocks. This type of control is rarely encountered at the present time, since for actuation in flight the cocks must be located in the cockpit in a place convenient for control. Direct control intended for operation on the ground (fuel draining) is used far more often.

Remote control makes it possible to control from the cockpit pumps or cocks located at a considerable distance and in locations with difficult access. This control may be electric, pneumatic, hydraulic, or mechanical. The command for actuating the remote control is given manually or automatically.

If we examine present-day manifolds for feeding fuel to the the engines when using the open system to vent the fuel tanks, we find that the devices which create the fuel flow are the boost and transfer pumps.

The design of the equipment for controlling the boost and transfer pumps depends on the type of pump drive. Pumps with electric motors are actuated by means of switches in the electrical circuit. Pumps with pneumatic and hydraulic drive are actuated by creating pressure in the supply line and connecting this line with the pump. The latter operation is achieved by actuation of a cock. The emergency turbopumps which are airstream driven require that the turbine be extended into the airstream, which is achieved by pneumatic control or some other technique.

Electric control of pumps and cocks is widely used. The low weight of the control lines, fast response, and possibility of automation are the advantages of this type of control. The disadvantages include fire hazard and dependence on electric power sources.

When manual control of a fuel system component with an electric drive mechanism is used, switches are used to apply current to the electric motor, which rotates the pump shaft or performs the required operations to actuate the cock.

The large number of boost and transfer pumps on the modern airplane and the necessity for following a definite program of fuel feeding from the tanks have led to automatic control of the pumps. Sensors of the automatic control system are located in the fuel tanks for this purpose. When they are actuated, they send a signal to relays, which then direct current to the controlling contactors.

The sensitive element of the automatic control for the fuel sequencer with electric actuator is a float-type inductive sensor, consisting of a coil and a float. The coil is hermetically sealed and has an open magnetic circuit. The float has an iron base — a sleeve which serves to close the coil armature. When the float descends into the sensor coil field the magnetic circuit closes and changes the inductance of the coil, which is connected to an electrical bridge. The bridge balance is disrupted and a potential difference appears across the ends of the diagonal and is directed to the excitation winding of a high-sensitivity relay. The latter actuates and its contacts close or open the power circuit to the boost pump contactor winding. Control of the cocks for supplying fuel or air to the pumps with hydraulic and pneumatic drive can be performed similarly.

If the sensors are located in the fuel tanks in the sequence corresponding to the fuel usage program, and if the sensors, relays, and pumps are connected into a single system, we obtain a system for automatic fuel sequencing.

Pneumatic control is quite safe with regard to fire hazard but leads to considerable plumbing weight and has large lag. Therefore it is usually used as an emergency control in case of electric power failure. It has been widely used for in-flight fuel dumping manifolds, when compressed air opens the dump cocks and valves, and has also been used to extend pump turbines into the airstream.

Hydraulic control is basically similar to pneumatic control. However, if the aircraft engines and their compressors are not operating use of pneumatic control is possible if there is pressure in the ship's system bottles or from ground power units, while in the same cases hydraulic control is possible only from ground power units. Hydraulic control is used for regulating fuel transfer into the service tank.

Mechanical control is sometimes used for manual remote opening of a cock. The cock is actuated by means of cables or push-pull rods. Such control can be realized if the controlled component is not located far from the cockpit. Drawbacks of this type of control are friction of the transmission cables on the pulleys, bending of the rods, the large force required to actuate the cock; this type of control is very rarely encountered today.

Operational Monitoring Equipment

Monitoring of fuel system operation consists in providing the crew with the required information on control actuation and system status. One of the objectives of this monitoring is to verify pump operation and cock position. Another monitoring objective is to assess the system status with regard to the basic parameters: fuel pressure, fuel flowrate and fuel remaining.

To provide monitoring of fuel system operation there are indicators for cock positions, fuel pressure, fuel quantity remaining, pressure gages, fuel flowmeters, and fuel quantity gages.

The position of direct-control cocks is determined from the position of the controlling lever. Electrical indication is provided for cocks which are remotely controlled. Limit switches are mounted on the cock and after its opening or closing they direct a signal to a light of prescribed color. Usually cock opening is indicated by a white or yellow light, closing by a red or blue light. However, sometimes the opening of certain cocks, the crossfeed for example, is indicated by a red light.

Pump activation is verified by the pressure differential created, which is determined by pressure gages. In many cases the exact value of the pressure is not of any interest, and it is only necessary to be sure that the pressure corresponds to the permissible range. Then pressure switches are installed and connected by an electrical circuit with a light. Actuation of the pressure switch makes the light come on (green light) or go out (red light). At the present time use is made of standardized pressure switches (SPS and SPSA) which respectively open or close the electrical circuits after the given differential pressure is reached in the manifold.

Fuel quantity gages and fuel flowmeters are used to measure the fuel quantity. The existing flow quantity gages are divided into float and nonfloat types, depending on the principle used in the pickup unit. The indications of the float type correspond to the quantity of fuel of a single grade at a definite temperature in level flight. During aircraft maneuvers such fuel quantity gages may give very large errors, since inertia forces act on the float and the fuel mass. If the fuel density varies, the degree of indication error varies. Therefore change of the fuel grade and temperature leads to different indications.

The nonfloat fuel quantity meters may be electrical-capacitance, ultrasonic, or isotopic. The electrical-capacitance types are most widely used. Their operating principle is based on transformation of a nonelectrical quantity (varying fuel level in the tank) into an electrical quantity — electrical capacitance, with the aid of capacitive transducers. With variation of the fuel level in the tank there is a proportional change of the capacitance of the condenser, since the dielectric constant of air is approximately half that of the fuel (kerosene). The condenser capacitance change causes a potential difference to appear in the bridge diagonal, which is transmitted to an electronic amplifier and the indicator. The indications of the fuel level transducers are also transmitted to the low-fuel-level warning lights.

The electrical capacitance fuel level gages are simple in construction and have high reliability. Moreover, they can be used to measure the fuel weight, since the dielectric constant of the fuel depends on its density. However, the error of these fuel quantity gages in flight is comparatively large, since the inertia effects on the fuel mass lead to variation of the fuel level in the tank.

Flow meters may be totalizing or instantaneous. The former measure the fuel quantity supplied to the engine from the moment the engine is started or the flowmeter is activated. If the indicator of the totalizing flowmeter is set to a position corresponding to the quantity of fuel serviced into the portion of the fuel system being metered (this can be done from the indications of the fuel quantity gage), then the difference between the serviced fuel quantity and the quantity used indicates the fuel remaining. Consequently, the totalizing flowmeter duplicates the indications of the fuel quantity gage. This facilitates troubleshooting. Since the flowmeter indications are independent of aircraft attitude and maneuvers, it is a more accurate instrument than the fuel quantity gage. The instantaneous flowmeters indicate the magnitude of the fuel flow per unit time. This makes it possible for the crew to judge the engine operating regime. Such flowmeters are often used together with the totalizing meters.

With respect to sensor operating principle, flowmeters may be of various types, but the most widely used are the velocity meters, in which the rate of rotation of an impeller located in the stream is proportional to the liquid velocity and the corresponding flowrate, which makes it possible to determine both the instantaneous flowrate and the total fuel consumption.

Operation

During fuel system operation the following basic functions are performed: servicing the tanks, draining the fuel, and maintenance of the fuel system.

Servicing the fuel tanks is an important, laborious, and lengthy operation in preparing the aircraft for flight, and in certain cases determines the time required to prepare for flight. Operational reliability of the fuel system components depends on the fuel quality (absence of water, mechanical particles, and so on). Problems in fuel system operation have a direct effect on engine operation and flight safety. The fuel quality is checked when it is delivered to the field, during storage in stationary reservoirs*, right on the field prior to servicing, and after servicing the aircraft.

Fueling is accomplished with the aid of stationary or mobile equipment, which must have the proper filtering, water separating, intake, and delivery equipment. Personnel of the airfield servicing crew verify the technical condition of the fueling equipment before beginning servicing from hydrants or prior to departure of mobile fueling trucks from the fuel storage area. Prior to pumping fuel from the storage reservoirs to the delivery hydrants, or prior to filling the tankers, checks are made of the fuel quality in accordance with the requirements of government standards. The fuel is certified on the basis of laboratory analysis.

Prior to beginning the operation of the delivery hydrants or on arrival of the tankers on the field, the fuel quality and the condition of the fueling equipment are checked by the servicing crew shift foreman. During the field inspection a check of the certificate is made to be sure the fuel corresponds to the specifications and operating instructions for the given aircraft type, and also the instructions on use and inspection of fuels, lubricants and special fluids, and the condition of the fueling equipment and filters is checked. After this a fuel sample is taken from the delivery hydrant or from the tanker sediment drains (after it is allowed to stand for at least 15 minutes) and a check is made that the fuel is clean. The sample or sediment is poured into a clean glass vessel in order to detect the presence in the fuel of water, ice crystals, or mechanical particles.

Water may be present in the fuel in three forms: Free, emulsified, and dissolved. When fuel containing any free or emulsified water is agitated, the acids and alkalis present in the fuel dissolve

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*Translator's Note: The Russian word used here is "laboratory", but reservoir seems to be more correct.

in the water and the fuel becomes turbid. More effective is the determination of these contaminants using potassium permanganate ($KMnO_4$). After several grains of $KMnO_4$ are dropped into the fuel it will take on a violet color if there is free or emulsified water present. To detect dissolved water, use can be made of a composition consisting of one part copper sulfate ($CuSO_4$) and two parts potassium iodide (KI). If water is present, after several grains of this composition are dropped in the fuel it will take on a yellowish color.

If the results of the field check of the fueling equipment and the fuel quality are positive, the inspector signs the certificate permitting fueling to proceed. Prior to fueling the flight engineer or the aviation technician checks the fuel grade required for the given engine, the presence on the certificate of the servicing crew supervisor's signature, the condition and cleanliness of the delivery nozzles, readiness of the fuel system to receive the fuel, and then takes the necessary safety measures (grounding the aircraft and the fueling units, taking the usual fire protection precautions).

Each aircraft type has its own specific fueling order, which is a function of its center of gravity location and maximal unloading of the wing in flight. Usually the tank fueling sequence is the reverse of the fuel usage sequence. Considering the possible fuel expansion if heating takes place, part of the tank is left unfilled. The serviced fuel should be allowed to stand for no less than ten minutes, after which the sediment is drained from the aircraft tanks or groups of tanks and the fuel cleanliness is checked. This same operation is performed during the preflight inspection.

If it is necessary to drain the fuel from the tanks on the ground, the vessels used for draining are connected with the tank drain cocks by means of flexible hoses. The boost and transfer pumps can be activated and the drain vessel can be connected with the manifold drain cock to accelerate fuel draining. The fuel drain sequence on the ground should correspond to the fuel usage sequence in flight.

__Maintenance__ of the fuel system on the ground and in flight involves performing all the operations specified in the maintenance regulations and the flight operations and pilot's manuals for the given aircraft type. All the components and connections must be inspected carefully. Particular attention is devoted to the absence of fuel leaks, since they lead to significant loss of fuel and may cause fires. Vibration, corrosion, deformation of fuel system elements under dynamic loads, poor assembly, and loose connections can be factors leading to loss of sealing of the lines and components. The mountings of the components and the connections must be checked and if necessary steps must be taken to eliminate any leaks.

Flushing of the fuel filters is one of the most frequently performed periodic operations. At low temperatures the moisture present in the fuel (particularly in kerosene) crystallizes and deposits on the filter walls, which leads to reduction of the fuel flow capacity and even termination of fuel flow. Improvement of the fuel filterability at low temperatures is achieved by the use of additives, which prevent separation of ice from the fuel. In the wintertime it is desirable that the fuel tanks always be filled when the aircraft is on the ground, since when flexible tanks are not completely full frost forms on their walls and can then get into the fuel and clog the filter.

At high temperatures (above 100° C) the existing grades of kerosene oxidize intensely, as a result of which insoluble sediments and resins form in the fuels and can clog the fuel filters and disrupt normal operation of the fuel system components. Microcontaminants present in the fuel have an influence on the formation of sediments. Improvement of the prefiltering on the ground leads to a considerable reduction of sediment formation.

Cleanliness of the vent line external fittings must be carefully checked. Clogging of the vents can occur as a result of entry of dirt, mud, snow into the vent or because of freezing.

During engine operation a check is made of the operation of cocks, boost and transfer pumps, fuel pressure, transfer, and low-level warning lights, pressure gages, fuel quantity gages, and flowmeters.

If the boost pumps fail (for example, if the electric system stops) flight is possible at moderate altitudes (up to 8000 m) without developing load factors on the airplane, at cruising power, and with smooth movements of the engine power lever.

In case of shutdown of some engines or failure of the boost pumps in one section of the manifolds which provide the fuel supply to the engines, if there is a crossfeed valve it is used in accordance with the fuel system operating instructions.

A large increase of the fuel consumption on the fuel quantity gage without the same indication on the flowmeter indicates a leak in the segment from the tanks to the flowmeter. A considerable difference in the fuel flowrate of one of the engines, observed on both the fuel quantity gage and the flowmeter (if this is not an instrument error), indicates fuel leakage in the segment from the flowmeter to the engine. In this and other cases steps are taken as specified in the flight operations handbook and pilot's handbook for the given aircraft type.

The boost pumps are turned off only after the engine rotor comes to a complete stop to eliminate the possibility of air lock formation and pump seizure because of the absence of lubrication (by the fuel) of the engine pump elements.

CHAPTER 4

OIL SYSTEMS

Powerplant operating reliability depends to a considerable degree on the lubrication conditions of the engine rubbing surfaces and adequacy of the heat removal from the engine assemblies and parts. Lubrication of the moving engine joints is necessary to reduce friction and wear of the parts, protect them against corrosion, remove the heat released during friction, and also to carry away the solid particles which are formed between the rubbing surfaces. Failure of the oil supply, even very brief failure, leads to engine overheating, damage to the engine bearings, binding of the gas turbine engine rotor or breaking of piston engine connecting rods.

The oil in powerplants is also used as the working fluid of various automatic devices: mechanisms for changing propeller blade angle, fuel controllers, governors, and so on. The modern aircraft oil systems are exclusively of the circulating type. The systems include provision for filtering and cooling the oil and preparation for the next cycle. The grade of oil used is determined by the loads on the lubricated parts, their operating temperatures, and the type of bearings used.

Transition to supersonic flight speeds, increase of the bearing loads, and also increase of the turbine inlet gas temperatures lead to considerable increase of the temperature of the engine assemblies and parts. Under these conditions the low-viscosity mineral oils vaporize and oxidize markedly. The vaporization leads to increased oil viscosity and makes engine starting difficult at low temperatures. Oil oxidation leads to the appearance of a large amount of sediment. These factors have forced the use of synthetic oils to lubricate the engines of supersonic airplanes. The synthetic oils include many esters, silicones, and some other substances.

The engine lubrication system consists of two segments: external and internal. The external section is a component part of the powerplant equipment; the internal section is part of the engine itself. We shall examine the operation of the external segment of the oil system, since the arrangement and operation of the internal segment are examined in engine design.

Requirements

1. Provide reliable oil supply to the engine in all engine operating regimes over a wide range of flight speeds and altitudes and ambient temperatures.

The operation of the oil system is monitored on the basis of oil pressure and temperature. The pressure is used to assess indirectly the amount of oil entering the engine. For most engines the oil pressure at idling speeds is 2-2.5 kgf/cm^2, while at normal operating speeds it is 4-6 kgf/cm^2. The oil temperature characterizes the heat release rate of the engine parts. On the basis of satisfactory viscosity, the upper permissible temperature of the mineral oils is in the range 70-80°C at the engine inlet and 100-120°C at the outlet. The lower inlet oil temperature limit depends on the engine type and the oil grade used. Thus, the modern turbojet engines operating on low-viscosity oils can be started without preheating at ambient temperatures down to 25-30°C. In those cases in which high-viscosity oil is used, starting the engines at these temperatures without preheating is not permitted.

2. On an aircraft with several engines each powerplant must have its own independent oil system. This makes for a more compact system, reduces line lengths, and increases engine operating reliability.

3. Possibility of cooling the oil with minimal power expenditures on operation of the cooling systems. If oil radiators are provided in the system, special devices must be used to maintain the oil temperature in the specified limits automatically.

4. Possibility of accelerated oil warmup. The warmup time depends on the amount of oil circulating in the system. This time must not exceed the time for the engine to warm up prior to advancing to high-power operation.

5. Absence of oil loss through the vent and overfilling of the engine with oil in all engine operating regimes on the ground and in flight.

6. Exclude the possibility of oil leaking from the tank into an inoperative engine.

7. Adequate strength, vibration resistance, and sealing of the lines and their connections, and also low hydraulic resistance of the oil system components.

8. The oil quantity in the system must be sufficient to provide for maximal duration flight with the maximal possible engine oil consumption. For aircraft with piston and turboprop engines there must also be an oil reserve for feathering the propellers.

9. Provide minimal oil consumption in flight. The oil consumption of the gas turbine engine is due primarily to loss of the light fractions through the breather into the atmosphere and oil leakage through the seals. In the piston engines oil is also burned in the cylinders.

For the modern gas turbine engines the oil consumption is low and amounts to about 1.5-3 kgf/hr. In the piston engines the oil consumption depends on the engine power and lies in the range of 10-15 grams/hp·hr.

10. Provide reliable removal of mechanical particles and gases from the oil. This is accomplished by the use of filters and air separators in the oil system. The filtering elements must have the filtering capability defined by the specifications for the given engine type and a contaminant capacity sufficient for operation without cleaning for the time period specified in the maintenance regulations. The filter design must provide special valves which bypass the oil through the filter without prefiltering in case of premature clogging of the filtering element.

11. Fire safety, which is provided primarily by tightness of the system. Moreover, the modern gas turbine engines are equipped with special systems which supply an extinguishing agent into the engine (into the reduction gearing case and the oil passages) in case of powerplant fire.

12. During filling of the system with oil, air locks must not form in the lines and components, since they can lead to engine failure.

13. Provide free access to the individual components, the possibility of measuring the oil quantity in the tank on the ground and in flight, fast servicing and complete draining of the oil from the system. The oil system must not interfere with the conduct of inspection, installation, and removal operations.

14. The oil system lines and fittings must be colored brown with arrows to indicate the oil flow direction.

Oil Operating Conditions

One characteristic of piston engine construction is the presence of rubbing surfaces of large area, on which the specific pressures amount to 250 kgf/cm^2 or more (in sliding bearings, for example, the pressures reach 600-800 kgf/cm^2). This leads to high heat transfer into the oil, which leads to the necessity of high oil flowrates through the engine (50-70 liters/min).

Oils with viscosity of 20-22 cSt at 100°C are required in order to ensure liquid lubrication of the friction joints under conditions of high specific pressures and high temperature, and also to provide good sealing of the clearances between the piston and cylinder.

If the viscosity is not sufficient, a liquid oil layer may not be retained in the clearances and this leads to intensified wear of the parts. Moreover, the oil bypass into the cylinders increases, with a resulting increase of oil consumption and intensification of scale formation. In connection with the high temperature conditions in the piston group zone during operation of the piston engine, conditions are created which are favorable for intense lacquer formation. The appearance of lacquer on the inner and side surfaces of the piston causes deterioration of its thermal conductivity and piston ring sticking. The loss of ring motility leads to more intense penetration of oil into the combustion chamber and of the hot gases from the chamber into the crankcase. Because of this there is a decrease of the engine power, increase of the engine temperature, contamination of the oil, possible breakage of piston rings, burning and sticking of the pistons, and connecting rod failure.

Excessively high oil viscosity is also undesirable, since this increases the friction forces and causes increased power loss during engine operation. At low temperatures high oil viscosity makes starting of a cold engine difficult, and after starting the oil flow is slow and oil spray formation deteriorates. Fresh oil does not reach the friction joints quickly enough and the oil which is in the clearances is heated by friction and flows away. Oil starvation, increased wear, and binding of the engine friction surfaces take place.

The MS-20 and MK-22 petroleum base oils have been found most acceptable for ensuring reliable lubrication of the piston engine under these severe operating conditions. These oils mix with one another in any proportions.

Characteristic of turbojet engine lubrication is the fact that the oil is not in direct contact with the combustion zone, as it is in the piston engine. The combustion chamber walls of these engines are not in contact with the lubrication system; therefore there is no burning of the oil and its consumption will be determined primarily by the oil losses through the engine breather system.

In these engines the primary form of friction is rolling friction. Therefore the power expended in overcoming the friction forces is not large. The specific pressures are much lower in turbojet engine bearings than in the piston engine, and therefore low-viscosity oils can be used (MK-6, MS-6, MK-8). There is no need to circulate a large quantity of oil to remove heat from the bearings, since the amount of heat released is small.

In addition to the bearings, in the TJE the oil provides lubrication for the geared accessory drives, where the oil operating conditions are comparatively light because of the low specific pressures in the gears and the low sliding speeds.

In some TJE designs the front compressor bearing is located in the diffuser inlet duct. Its temperature may be below freezing, while the turbine bearings are heated by heat transfer to them from the engine hot section.

The turbine bearings are heated not only during engine operation but also after engine shutdown, when oil flow for cooling the bearings and air flow for cooling the turbine are not available, and the heat from the engine hot section is communicated to the bearings for some time period. This circumstance must be taken into account in selecting the oil for lubricating gas turbine engines.

Turboprop reduction gearing transmits a large amount of power and the gears require abundant lubrication. This leads to the use of high-viscosity oil. Problems in using high-viscosity oils are reduction of the propeller blade angle rate of change, difficulty in starting the engine at low temperatures, reduction of fuel controller regulation sensitivity, and so on. In this connection there are oil systems in which the reduction gearing is lubricated by high-viscosity oil while the engine is lubricated with low-viscosity oil.

The use of separate oil systems complicates the engine construction; therefore a single oil system is usually used and a mixture of oils with high and low viscosities is selected for lubrication. With increase of the engine power or with increase of the specific pressures on the gear teeth, the amount of high-viscosity oil in the mixture must be increased.

Techniques for Removing Air Inclusion from Oils

The primary cause of disruption of normal oil system operation is saturation of the oil with air. Under normal atmospheric conditions the oil contains 8-10% by volume of air in solution. Atomization of the oil during lubrication of the bearings and numerous drives and transmissions leads to mixing of the oil and air. The result is the formation of an air/oil emulsion, which is then drawn out of the engine by the scavenge pumps. For reliable removal of the oil from the engine sumps the output of these pumps must be greater than that of the pressure pumps (Figures 91 and 92). This further saturates the oil with air, since the pumps draw out a large amount of free air and gases in addition to the emulsion. The air which is thus mixed with the oil is in the suspended state in the form of bubbles. In the components of the scavenge circuit (pumps, radiator)

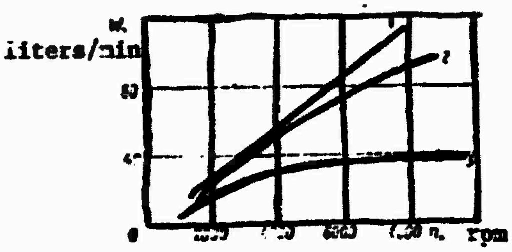

Figure 91. Output of OMN-30 oil pump assembly.
1,3 - pressure stage with relief valve blocked and adjusted; 2 - scavenge stage

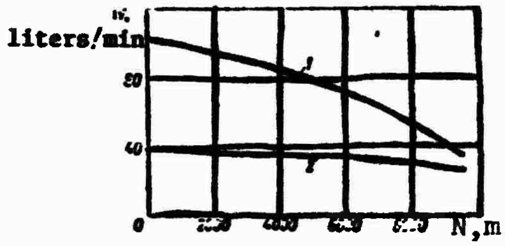

Figure 92. Output of OMN-30 oil pump assembly. 1 - scavenge stage; 2 - pressure stage with relief valve adjusted.

the air bubbles are broken down and part of the air is further dissolved in the oil because of the high pressure in the scavenge circuit. The large bubbles rise as the oil enters the tank. Since the pressure in the tank is lower than in the scavenge line, the dissolved air separates from the oil in the form of very fine bubbles. However, the rate of rise of these bubbles is small and they are not able to escape into the tank air space.

Studies [20] have shown that for a constant bubble diameter its rate of rise is inversely proportional to the oil viscosity, while the rise time is directly proportional to the viscosity (Figure 93) and the height of the oil level in the tank.

The oil residence time in the tank has a significant effect on the amount of air which gets into the system suction line. It is obvious that only air bubbles whose diameter is such that their rate of rise is greater than or equal to the oil lowering rate can escape into the tank air space. In the single-contour oil systems of turbojet powerplants the rate of oil lowering rate in the tanks lies in the range of $(3\text{-}5)\cdot 10^{-3}$ m/sec. Calculations made for the MK-6 oil (Figure 94) show that for these lowering rates and an oil temperature in the tank of 70°C the minimal diameter of the rising bubbles will be 0.14-0.18 mm. Air bubbles of smaller diameter cannot rise to the surface, and they flow together with the oil to the pump.

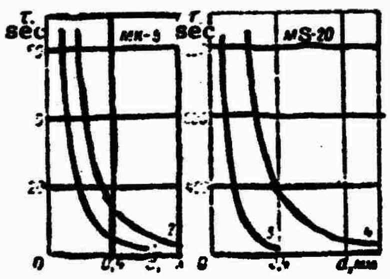

Figure 93. Time required for separating air versus bubble diameter for oil temperature 50°C (curves 2,4) and 100°C (curves 1,3).

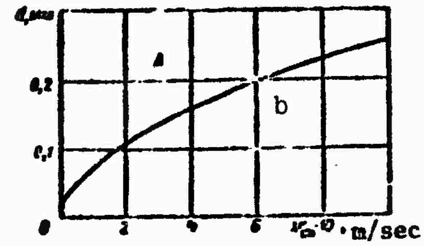

Figure 94. Boundary between diameters of air bubbles which rise (zone A) and remain in the oil (zone B) as a function of oil settling rate in tank.

Thus the amount of air entering the suction line depends on the oil grade and temperature, pressure in the scavenge manifold, oil residence time and level in the tank, and the size of the air bubbles. The more air there is pumped from the engine along with the oil, the more air there will be in the suction line. With increase of the pressure in the scavenge line and reduction of the oil viscosity, the air solubility in the oil increases and therefore the smaller is the fraction of the air which escapes during the time the oil stays in the tank.

With increase of the flight altitude the pressure in the tank and at the pressure pump inlet decreases. The volume of the air bubbles present in the oil increases and the amount of oil entering the pump decreases. Moreover, the presence of the air bubbles increases the line hydraulic resistance.

Special measures must be taken to reduce the percentage air content in the oil. Thus, in order to reduce mechanical breakup of the oil and its mixing with air it is advisable in the design of the internal portion of the engine oil system to locate the sumps and oil drain paths in regions remote from the rotating parts. Since increase of the difference of the scavenge and pressure pump outputs leads to increase of the air content in the oil, this difference should not be made too large. This condition cannot always be satisfied; however, the ratio of the output of the scavenge pumps to that of the pressure pumps can be reduced to 2.5-3 by rational location of the pumps, sumps, and oil drain paths.

Centrifugal air separators — centrifuges — are used to ensure the most complete removal of air from the oil. With a properly selected centrifuge the air content in the oil can be reduced to 4-5%. Among the other techniques which aid in reducing the air content in the oil, one worthy of special attention is the use of special additives (usually of silicone origin) which reduce the surface film strength, intensify the coagulation of fine bubbles, and make it easier for the bubbles to rise to the surface.

Airplane Powerplant Oil System Configurations

Two types of oil systems are most widely used at the present time: single-contour and double-contour. The first are used primarily on aircraft with PE and TJE; the second are used mostly with TPE. The single-contour oil system (Figure 95) has the following oil circulation scheme: tank-engine-radiator-tank.

The line from the tank to the pressure pump is called the suction line. This segment must be as straight, short, and large in diameter as possible.

To improve the pressure pump operating conditions, which is particularly important during starting with cold oil, the tank should be located above the pump and as close as possible to the pump. Since filters in the suction line increase the line resistance markedly and thereby reduce the ceiling of the oil system, their installation here should be avoided, in spite of the advantages from the viewpoint of protecting the system against dirt and other mechanical particles which may get into the tank.

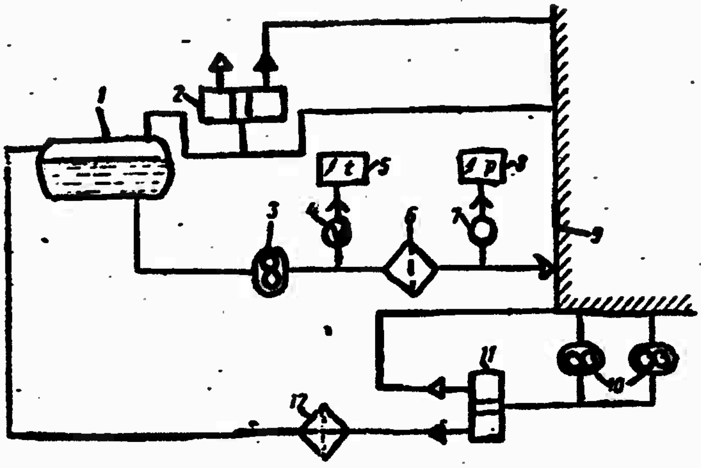

Figure 95. Schematic of single-contour oil system: 1 - tank; 2 - centrifugal breather; 3 - pressure pump; 4,5 - temperature sensor and indicator; 6 - filter; 7,8 - pressure sensor and indicator; 9 - engine; 10 - scavenge pump; 11 - air separator; 12 - radiator.

The oil system segment from the pressure pump to the scavenge pump is the internal engine circuit. In this segment the oil, after cleaning in the filters, provides lubrication for the engine rotor bearings, various drives, and in many cases is used as the working fluid in automatic devices. Additional cleaning of the oil is accomplished by filters installed ahead of the nozzles which direct the oil to the bearings, in the automatic regulators, and other accessories. Sometimes screens are installed at the oil exit from the engine to protect the scavenge pumps against entry of mechanical particles. The appearance of metal chips on the surface of these filters is usually the first indication of abnormality in engine operation.

A shutoff valve which opens at a pressure exceeding the oil static head at the pump inlet is frequently installed downstream of the pump to prevent oil leakage from the tank through the clearances of the pressure pump into an inoperative engine. Oil temperature and pressure sensors are installed in the internal engine circuit.

The portion of the system from the scavenge pumps to the oil tank is termed the scavenge circuit. This part of the system includes the radiator, air separator, and other devices used to prepare the oil for the next usage cycle. The centrifugal air separator is usually installed ahead of the radiator. This is done because the air separation process is facilitated in the hot oil, and also the radiator heat transfer is improved if the oil flowing through it has had the air inclusions removed. Moreover, the centrifuge creates a head sufficient to overcome the hydraulic resistances of the radiator and other components installed in the scavenge circuit.

On some powerplant types the oil flows to the compressor inlet case struts and to a special engine oil spacer prior to entering the radiator. In the struts and spacer the oil is partially cooled by the air entering the engine and is then directed to the fuel/oil radiator, from which after being cooled it enters the tank. On some aircraft types with turbojet engines fuel-oil assemblies are used. In the case of this assembly, which is at the same time the oil reservoir, there are usually installed the fuel-oil radiator, the fuel and oil filters, devices for oil pickup and providing breathing for the tank. This arrangement makes it possible to provide in a single assembly oil cooling, fuel filtration, and also continuous supply of oil to the engine pumps during various aircraft maneuvers and action of negative load factors.

The basic circuits of the double-contour oil system (Figure 96) are analogous to those of single-contour systems. The difference between these systems is that in the double-contour system there is a boost pump (makeup pump) and two oil flow contours: primary and secondary. Up to 90% of the pumped oil circulates through the primary contour (pressure pump-engine-scavenge pumps-centrifuge-radiator-pressure pump) and only 10% flows for heating through the secondary contour (radiator-oil tank-boost pump). In this case all the oil pumped through the engine passes through the centrifuge. The oil is directed through the secondary contour into the tank only after it leaves the radiator. The diameter of the orifice in the secondary

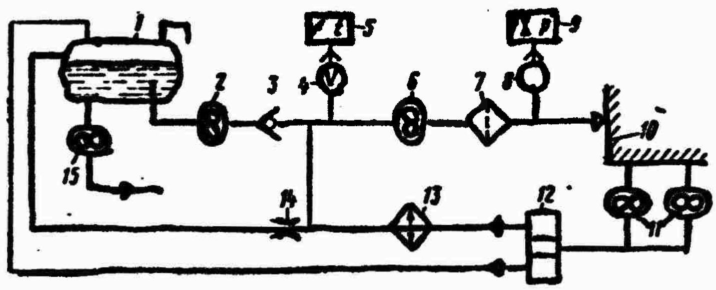

Figure 96. Schematic of two-contour oil system: 1 - tank; 2 - makeup pump; 3 - check valve; 4,5 - temperature sensor and indicator; 6 - pressure pump; 7 - filter; 8,9 - pressure sensor and indicator; 10 - engine; 11 - scavenge pumps; 12 - air separator; 13 - radiator; 14 - orifice; 15 - feathering pump

contour is selected to bypass the optimal amount of oil. This oil and the oil consumed during engine operation flows from the tank into the primary contour (to the inlet to the pressure pump) with the aid of the boost pump.

Since not all the oil in the tank circulates through the engine, its warmup is accelerated and the time required to prepare the aircraft for flight is reduced. A second advantage of this system is the possibility of obtaining a higher ceiling than for the single-contour system. Increase of the ceiling is obtained by installing in the suction line a boost pump, which maintains a definite differential pressure at the pressure pump inlet, which eliminates the formation of air locks at the pump inlet.

A modification of the double-contour oil system is the short-circuit system (Figure 97). In this system there is no orifice (see Figure 96) or line segment between the radiator and tank. In the short-circuit system the oil, bypassing the tank, circulates along the contour: engine-radiator-engine. The oil systems of the Il-18, An-10, and An-24 airplanes use this configuration.

The initial filling of the double-contour and short-circuit systems with oil, and also replacement of the oil consumed by the

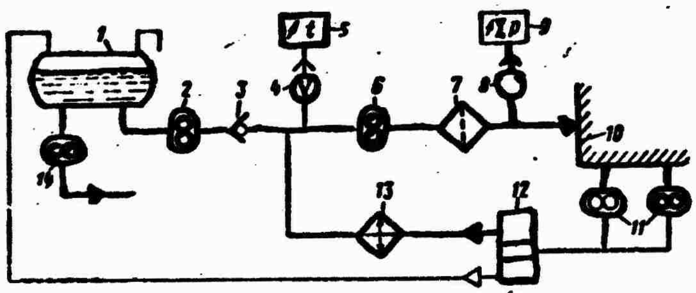

Figure 97. Schematic of short-circuit oil system: 1 - tank; 2 - makeup pump; 3 - check valve; 4,5 - temperature sensor and indicator; 6 - pressure pump; 7 - filter; 8,9 - pressure sensor and indicator; 10 - engine; 11 - scavenge pump; 12 - air separator; 13 - radiator; 14 - feathering pump.

engine, are accomplished from the tank by the makeup pump 2 (see Figure 97). The pressure pump 6 supplies oil through filters to lubricate the engine.

The return oil is directed by the scavenge pumps 11 to the air separator 12. After removal of the air the oil is directed to the radiator 13, and then after cooling it enters the oil pressure pump. The centrifugal air separator creates an oil head at the inlet to the pressure pump and backpressure at the oil outlet from the boost pump. When the oil pressure at the entrance to the pressure pump rises above 0.8 kgf/cm^2, the makeup pump relief valve opens and the excess oil is returned to the suction line. When the oil pressure at the inlet to the pressure pump drops, the makeup pump begins to replenish the system with the required amount of oil. The air separated from the oil in the centrifuge is directed into the tank. In the double-contour and short-circuit oil systems of airplanes with TPE the suction line pickup fitting is not located at the lowest point of the tank, but rather somewhat higher, which provides a reserve oil supply for feathering the propeller.

Properly designed tank venting must be provided to ensure normal operation of the oil system. Depending on the air separation technique used and its efficiency, the oil tank may be connected through the vent line to the atmosphere or to the engine breather system. Tank venting through the engine is widely used, since this eliminates oil loss from the tank into the atmosphere and reduces entry of dust and moisture into the tank.

Such systems operate particularly well when the tank is vented to the atmosphere through a centrifugal breather mounted on the engine.

Sometimes the oil tank vent line is connected to special oil recovery (vent) reservoirs, from which the air flows through the vent lines to the atmosphere and the oil drains into the engine sump or into the oil tank. The installation of such vent reservoirs is a good means for eliminating oil loss through the vent during negative load factor conditions. However, their use is limited because of the weight increase and the increased oil system complexity.

It is advisable that the oil tank vent lines be brought out to a warm area (for example, into the radiator duct) in order to avoid the danger of freezing. Relief valves are sometimes installed on the vent line inside the cowling in order to improve vent system reliability.

Regardless of the oil system configuration, it includes elements which provide for servicing, draining, and monitoring the oil quantity.

The oil lines are installed with adequate slope to the drain valve to provide for rapid draining of the oil. Particular attention is devoted to draining the oil from the engine sumps, oil tank, and oil radiator. In designing the drain devices steps should be taken to ensure that the oil does not come into contact with the electrical wiring, engine hot parts, and tires.

Characteristics of Helicopter Powerplant Oil Systems

Helicopter powerplants have two independent lubrication systems: one to supply oil to the engine and the other to lubricate the

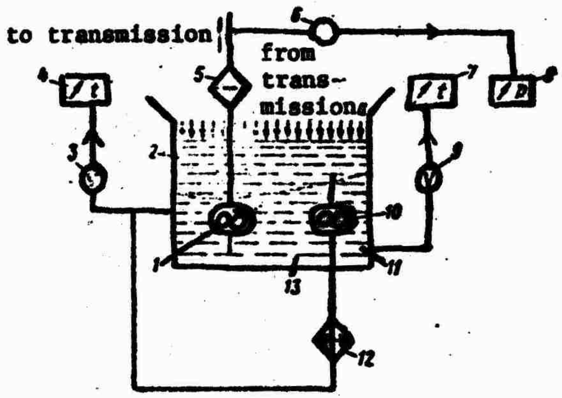

Figure 98. Main transmission oil system: 1 - pressure pump; 2 - "hot" compartment; 3,4 - oil-in temperature sensor and indicator; 5 - filter; 6,8 - pressure sensor and indicator; 7,9 - oil sump temperature indicator and sensor; 10 - scavenge pump; 11 - sump; 12 - radiator; 13 - "cold" sump compartment.

gearboxes. The schematic of the helicopter engine lubrication system does not differ in any essential way from the single-contour airplane powerplant oil system examined above. The MS-20 or MK-22 oils are used for helicopters with piston engines, while the MK-8 (MK-6, MS-6) oils are used for helicopters with gas turbine engines.

The main rotor transmissions have a high reduction ratio, since they reduce the engine rotor speed to the required main rotor speed. The transmission lubrication system has certain peculiarities in connection with this situation, and in connection with the type of transmission and the power transmitted to the main rotor.

If the power transmitted to the main rotor is not large, the lubrication system consists of a sump which acts at the same time as the oil reservoir, a pump, screens, and spray nozzles. There are usually no external oil lines or radiators. The oil is cooled in the transmission sump by air flowing around it from a special fan. Transmission of high power through the reducing gears makes it necessary to have external oil system components and the oil is cooled in air-oil radiators.

Transmission lubrication is accomplished as follows. The pressure pump 1 (Figure 98) picks up oil from the "cold" compartment 13 and supplies it to lubricate the transmission. The return oil flows from the transmission into the sump "hot" compartment 2, from which the scavenge pump 10 delivers it to the radiator 12 for cooling. The oil returns from the radiator into the "cold" compartment. System operation is monitored by the oil pressure in the pressure line and temperature in the "cold" compartment and the scavenge line. To increase oil system reliability the scavenge line pickup fitting is raised somewhat above the bottom of the sump, which ensures that a definite quantity of oil will remain in the sump if there is a leak in the scavenge line. The oil remaining in the sump provides normal lubrication of the transmission for the time necessary to select an emergency landing spot. The existence of a leak is indicated by increase of the oil temperature in the sump. In addition to the main transmission, in powerplants with gas turbine engines the transmission oil system also provides lubrication of individual engine components (for example, the free turbine transmission).

Lubrication of the main gearboxes can be provided by a mixture of MS-20 and MK-8 oils (for example, 75% MS-20 and 25% MK-8) and also by hypoid gear oil. The selection of the oil grade depends on the gearbox type, the magnitude of the specific loads in the gearing, and the power transmitted by the gearbox.

Autonomous lubrication systems with servicing of the oil directly into the gearbox case are used in the intermediate and tail gearboxes The friction surfaces are pressure or spray lubricated, and certain types of gearboxes (powerplants with gas turbine engines) use both forms of lubrication.

Supply of oil under pressure to the gearbox is accomplished by a pump, which together with a filter and relief valve is located in the gearbox oil assembly case. Splash lubrication is provided by having the rim of one or more of the gears partially submerged in the oil. As these gears rotate an oil emulsion is created in the case and provides adequate lubrication of the gearbox. Hypoid lubricants are used as the oil for the intermediate and tail gearboxes.

Oil Cooling Techniques

The oil temperature must be regulated in order to maintain the engine temperature within acceptable limits when its operating conditions are changed. A temperature difference of 35-50°C between the oil entering and leaving the engine is considered normal.

Various oil cooling techniques are used, depending on the type of aircraft and engine, the flight speed and altitude, and the engine heat release rate. In the TJE with relatively low thrust the amount of heat which must be removed from the engine is small, and therefore the oil is cooled directly in the oil cavities of the engine or in the tank.

With increase of the engine thrust there is an increase of the amount of heat to be removed, and this requires the presence of radiators in the powerplants. Fuel-oil radiators are used most frequently for this purpose. Cooling of the oil by the fuel is advantageous from the viewpoint of reducing the radiator size and weight, and also utilization of the heat given up by the oil to the fuel. Moreover, the use of the fuel-oil radiators avoids the expenditure of power to overcome the external drag of a radiator installation. When fuel-oil radiators are used there is no external system for regulating the oil temperature and it is automatically maintained in the specified limits as the engine operating conditions are changed as a result of the change of the amount of fuel flowing through the radiator and consumed by the engine.

Increase of the amount of heat removed from the engine (which is characteristic of the PE and TPE) makes it difficult to use fuel-oil radiators. This is caused by the fact that the heat Q_{req} which must be removed from the engine goes to heat the fuel in the radiator when fuel-oil radiators are used in the system, i.e.,

$$\Delta t_f = \frac{Q_{req}}{c_f \rho_f W_f} \text{ deg}$$

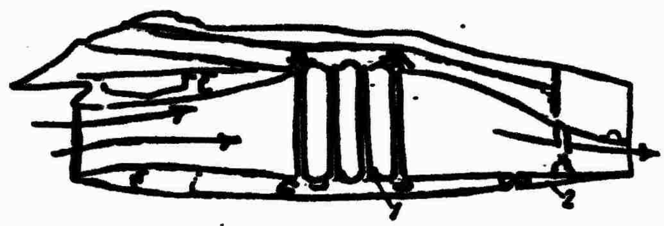

Figure 99. Installation of air/oil radiator in duct: 1 - radiator; 2 - flap.

where c_f is the specific heat of the fuel, J/kg·deg;
ρ_f is the fuel density, kg/m^3;
W_f is the fuel flowrate through the radiator, m^3/sec.

We see from this relation that the fuel temperature rise is directly proportional to the amount of heat removed from the engine and inversely proportional to the fuel flowrate through the radiator. Since, other conditions being the same, the fuel flowrates in the TPE are smaller and the amount of heat removed is greater than for the TJE, this can lead to excessive fuel temperature rise in the radiator. To avoid this situation use is made of air-oil radiators, or the fuel flowrate through the radiator is increased with subsequent return of part of the fuel to the tank. Since the second approach leads to complication of the fuel system design and increase of the output of the airframe boost pumps, which results in increase of the system weight, at the present time air-oil radiators are used on aircraft with PE and TPE. The use of air cooling is not feasible on supersonic airplanes because of the high stagnation temperatures.

In order to provide variation of the flow through the air-oil radiator, it is installed in a duct with variable exit area (Figure 99). To reduce the air velocity ahead of the radiator, the entrance segment of the duct is made in the form of a diffuser. The radiator is located in regions where the air is decelerated by the parts of the aircraft and which are swept by the propeller. This provides the required flow over the radiator in flight and during engine operation on the ground.

To avoid oil overheating, the air-oil radiators are designed for the most severe cooling conditions. However, there is also the danger of overcooling the oil under certain flight conditions. Such a case is that of extended gliding with throttled engines, when intense air flow through the radiator is combined with low heat transfer into the oil. Oil overheating is possible during engine operation on the ground and during takeoff, since in this case the intense heat transfer into the oil is accompanied by low air flow through the radiator. In the latter case special devices must be provided to increase the air flow through the radiator, for example, air bleed from the engine compressor into the radiator duct to create an ejector action (Figure 100), which is used on the Tu-114 and An-10 airplanes.

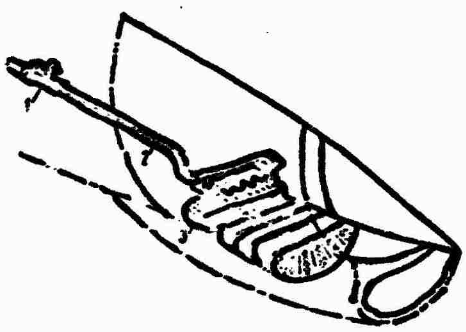

Figure 100. Air/oil radiator ejection schematic: 1 - valve; 2 - line; 3 - ejector.

The oil radiator duct flap serves to regulate the air flow through the duct by varying the exit area; control of the flap position can be automatic or manual. Automatic control is accomplished with the aid of special regulators, which maintain the oil temperature in the specified limits by varying the amount of air flow through the radiator honeycomb. The regulator sensitive element is a spiral bimetallic element, which is connected to a servo system. When the automatic regulation temperature is reached the contact-type servo system sends a signal to open the flap. A drawback of this system is the possibility of false actuation because of the poor vibrational resistance of the moving contacts.

Regulation systems are being developed in which resistance thermocouples are used as the oil temperature sensor and the comparison

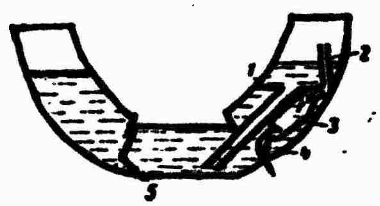

Figure 101. Oil tank with compartment for supply at negative load factors: 1 - baffle; 2 - vent tube; 3 - pickup compartment; 4 - delivery line; 5 - diaphragm.

element is a bridge in which one leg is the sensor. These systems are more reliable in operation because of the absence of moving contacts.

Components

The airframe part of the oil system consists of the tanks, lines, and radiators. The boost pumps, check valves, filters, pressure and scavenge pumps are engine components or are part of the internal engine system.

Tanks. Both flexible and metal tanks are used in the oil systems of modern aircraft. In construction and the requirements imposed on them, the oil tanks are similar to the fuel tanks. The flexible tanks are installed in containers fabricated from fiberglass or aluminum alloys. The tank shape is selected in accordance with the area in which it is installed.

Normal operation of the lubrication system depends to a considerable degree on proper placement of the pickup fitting in the tank. It should protrude into the tank to prevent entry of mechanical particles and moisture into the system. The oil should be returned into the upper part of the tank, since this increases the oil residence time in the tank and ensures that all the oil that is intended for this purpose will participate in the circulation. Inside the tank the end of the scavenge line tube is directed tangent to the wall so that the oil entering the tank flows along the wall without forming foam. It is recommended that the vent fitting on the tank be located as far as possible from the oil entry into the tank, since improper relative location of the return fitting and the vent line sometimes leads to excessive oil consumption. The filler neck with its screen is located at the top of the tank and the drain cocks are located at the bottom. A dip stick is provided on the filler neck or nearby. The transmitters for the remote oil quantity gages are located inside the tanks.

Special devices (horizontal baffles, pickup compartment, pivoting pickups, and so on) are installed inside the tanks to ensure reliable operation of the system under the action of negative load factors. One possible device of this sort is shown in Figure 101, and consists of the pickup compartment 3, separated from the main cavity by the horizontal baffle 1. During the action of negative load factors the oil is retained below the baffle, the ports of the pickup compartment remain in the oil, and therefore the oil supply to the engine is not interrupted.

Circulation wells are installed inside the tanks of some single-contour piston-engine oil systems. When the well is used a large part of the oil in the tank essentially does not take part in the general circulation. This results in accelerated oil warmup during engine starting, ensures adequate sensitivity of the system to control of the temperature regimes, and improves the lubrication quality. As oil is consumed the well is continuously replenished from the main tank volume. The foamy oil from the engine enters the well through the return line tangent to the inner surface with definite kinetic energy, as a result of which the oil flows down along a spiral path, clinging to the well wall, while the foam with the air bubbles is forced out into the central part of the well, from where it escapes the tank through the upper holes in the well.

Metal tanks are usually supported on steel straps which are tightened by turnbuckles. Special ribbing is provided on the tanks to increase the structural stiffness and prevent the straps from sliding. Strips of leather, rubber, felt, and so on are placed under the straps. When the tank is mounted on a lateral supporting surface (firewall, engine frame), a support is provided under the tank and the tank is strapped to this support. The flexible tank containers are bolted to supporting surfaces or the airframe structure. Rubber dampers are provided in the support fittings to reduce vibration transmission to the tank. After fabrication the oil tanks are subjected to thorough testing under vibratory, inertial, and hydraulic loads.

Radiators. Their construction is characterized by lightweight tubular structural profiles, reinforced by baffles. The spherical radiator shell is located between the profiles. This structural scheme permits increasing the pressure in the radiator to 4-5 kgf/cm^2 with a four-fold safety margin. All the structural elements, fittings, and cooling elements are usually unitized, which improves the fabricability of the parts and their reliability.

Structurally, the oil radiators are fabricated from an assembly of copper or brass tubes 250-300 mm long, 0.1-0.2 mm thick, and diameter 4-5 mm, enclosed in a brass housing. The tube ends are expanded into a hexagon over a length of two to three diameters. Radiators made from such tubes are termed honeycombs. The ends of the tubes are brazed together so that slots through which the hot oil circulates remain between the tubes. In the air-oil radiators cooling air circulates inside the tubes; in the fuel-oil radiators the fuel circulates in the tubes.

To increase the oil flow velocity (in order to improve the heat transfer and reduce the size of the stagnant zones), the radiator intertube space is divided into individual sections. Each baffle has ports through which oil flows from one section to another, passing through all the sections and changing its direction each time (Figure 102). It is for these reasons that the fuel cavity of the fuel-oil radiators is divided by baffles.

To prevent radiator honeycomb failure at low oil temperatures, relief valves are installed in the oil system lines or in the radiator itself, which permit the oil to bypass the radiator honeycomb or flow

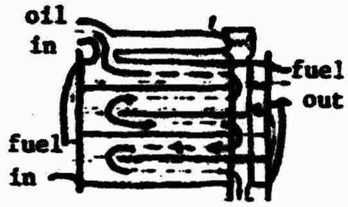

Figure 102. Oil and fuel circulation in radiator: 1 - relief valve

through the honeycomb, depending on the oil viscosity. When the viscosity is high (engine starting at temperatures below freezing), the radiator honeycomb hydraulic resistance increases and the pressure at the radiator inlet increases. The relief valve opens at a definite pressure and connects the radiator inlet and outlet cavities. The relief valves facilitate faster oil warmup, since the oil bypasses the radiator. As the oil in the system warms up and the pressure decreases, the valve connects the radiator into the system.

The radiator shape may be quite varied, depending on its location. Circular and horseshoe shapes are most common. The fuel-oil radiators are mostly circular, while the air-oil radiators are usually the horseshoe type.

Aluminum oil radiator designs have been developed. They first found application in systems for cooling helicopter powerplants. The radiator cooling elements are flat tubes, which are ribbed on the inner and outer sides by turbulizing plates. Characteristic of these radiators is their light weight and high thermal efficiency.

Oil Lines. Metal lines and flexible hoses are used in the external portion of the oil system. The metal lines located in the engine compartments are frequently fabricated from steel tubing; those in the other parts of the system are made from aluminum alloys. The lines are connected with one another by flexible couplings with two clamps on each end. In areas subject to vibrational loads, flexible hoses are used and their ends are terminated in standard hose fittings, which provide reliable interconnection of the hoses and also connection with the fittings on the system components. The flexible hoses are made from oil-resistant rubber and are covered with two layers of insulation (inner layer of AT-7 asbestos fabric, outer layer of ANChM fabric).

Flexible compensators, made in the form of a thinwall metal or Teflon shell with wire braid around its outer surface, are used in certain areas to ensure reliable operation of the lubrication system. These compensators can be in the form of a separate hose or an elastic insert. The latter make it possible to install lines with misalignment of the axes, and also make it possible to compensate for thermal displacements without any system leakage. The primary problem with flexible hoses is that they are heavier than metal lines and have more hydraulic resistance.

Depending on the magnitude of the oil flowrates, the diameters of the suction and scavenge lines used lie in the range of 25-40 mm, while the oil tank vent lines are in the 15-25 mm range. In running the lines care is taken not to form loops in which oil can be retained when the system is drained. In the wintertime this could lead to the formation of oil blocks, which cause line failure and engine oil starvation. Line disconnects are provided at the firewalls and at the individual system components (tanks, filters, pumps, valves) for convenience in engine installation. The connections of the lines to the pumps are very critical areas, since even the smallest leaks in these areas lead to air entrainment. The points where the lines are anchored are selected so that the natural frequency of the lines when filled with oil do not coincide with the exciting frequency. Usually the distance l between supports is chosen so that $l \leqslant 120\sqrt{d}$ for aluminum lines and $l \leqslant 150\sqrt{d}$ for steel lines, where d is the line diameter in millimeters.

Analysis

Capacity Determination

The oil system capacity E_{os} is calculated using the formula

$$E_{os} = E_{ot} + E_{line} + E_{eng} + E_p,$$

where E_{ot} is the amount of oil in the tank;

E_{line} is the amount of oil in the system lines and components;

E_{eng} is the amount of oil in the engine;

E_p is the amount of oil in the propeller cylinder group.

The amount of oil in the tank includes the quantity E_{con} consumed during flight and the minimal permissible nonexpendable oil quantity E_{min}.

The amount of oil expendable during flight

$$E_{con} = q_o \tau_{max} Kh$$

where q_o is the hourly oil consumption, kgf/hr;

τ_{max} is the maximal flight duration of the aircraft with the fuel system fully serviced, hr.

The hourly oil consumption is determined experimentally for each engine type and is presented in the corresponding manuals. The maximal flight duration is taken from the flight operations manual, or in the absence of these data is calculated from the formula

$$\tau_{max} = \frac{G_f}{mq_f}$$

where G_f is the total fuel supply on board, kgf;

m is the number of engines on the aircraft;

q_f is the engine hourly fuel consumption at the power for which the maximal permissible oil flowrate is given in the manual, kgf/hr.

The unexpendable oil quantity in the tank corresponds to that oil quantity which provides the specified oil pressure in all engine operating regimes and aircraft flight altitudes. For the TPE the minimal permissible amount of oil also includes the reserve (unusable) oil supply for feathering the propeller. The value of E_{min} can be found from the condition for obtaining the minimal required oil residence time in the tank which will provide stable operation of the system. E_{min} can be less when a centrifuge is used than when air separating devices are not used. In the latter case it is recommended that E_{min} be close to the magnitude of the oil flowrate through the engine. Usually E_{min} is determined experimentally for each engine type and is specified by the manufacturer.

The amount of oil in the system and in the propeller cylinder group can be measured or calculated on the basis of the system schematic, component data, and line sizes. The amount of oil in the engine can be found by draining the engine. For approximate calculations, assuming that the system operates using the dry sump principle, this oil quantity can be taken equal to zero.

To prevent oil loss through the vent system and to create better air separation conditions, the oil tank volume should be 20-30% more than the maximal permissible oil volume in the tank. The tank shape as well as its volume plays an important role. When a tall tank is used the cross-section area is smaller and the rate at which the oil descends in the tank increases, on the other hand the time available for air separation increases. When the tank horizontal dimensions are increased the oil descent rate decreases and air bubbles rise to the surface more easily. However, excessively small tank height leads to the danger of uncovering the pickup fitting during aircraft maneuvers and makes it difficult to warm up the oil in the tank.

Required Oil Flowrate Through Engine

The oil flowrate through the engine depends on the amount of heat Q_{req} which must be transferred into the oil and the permissible oil temperature rise. The engine specifications usually indicate the oil pressure at idle and full power conditions; however, sometimes only a single value of the pressure for a wide range of engine operating conditions is specified. This specification essentially determines the oil flowrate through the engine under the various engine operating conditions.

The oil flowrate through the engine

$$W_o = \frac{Q_{req}}{c_o\ q_o\ \Delta t_o}\ m^3/sec$$

where c_o is the specific heat of the oil, J/kg·deg;

Δt_o is the difference of the engine oil in and out temperatures, deg.

The amount of heat which must be removed from the engine by the oil is made up of the heat released as a result of mechanical losses in the bearings and reduction gearing, and the heat transmitted from the engine hot section, i.e.,

$$Q_{req} = Q_{red} + Q_{bear},$$

where Q_{red} is the amount of heat which must be removed from the reduction gearing and the various drives;

Q_{bear} is the amount of heat which must be removed from the bearings and engine hot section.

The heat transfer into the oil depends on many factors, the primary ones being the engine type and power, oil temperature, and ambient temperature. The heat transfer into the oil increases with increase of the engine power (Figure 103). Thus, for the Ai-20 engine at $N = 0.4\ N_{rat}$ and an oil-in temperature of 70°C, Q_o = 48 KJ/sec. When the engine power is increased to rated, the heat transfer increases to 60 KJ/sec. For a constant power the heat transfer decreases (Figure 104) with increase of the oil-in temperature because of the reduction of the temperature differential between the engine hot parts and the oil. Decrease of the heat transfer also takes place with reduction of the air temperature, which is a result of increased heat transfer from the engine to the ambient medium.

The amount of heat transmitted to the bearings from the hot parts depends on the efficiency of the cooling system and is not amenable to calculation because of the influence of a large number of factors.

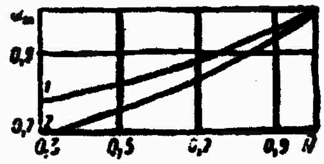

Figure 103. Heat transfer into oil when engine operating regime changes

$$\left(\bar{Q}_o = \frac{Q_o}{Q_{o:rat}};\quad \bar{N} = \frac{N}{N_{rat}}\right)$$

1 - engine power 4000 hp; 2 - 12,000 hp

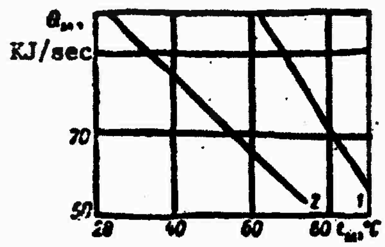

Figure 104. Heat transfer versus oil temperature: 1 - engine power 12,000 hp; 2 - 4000 hp.

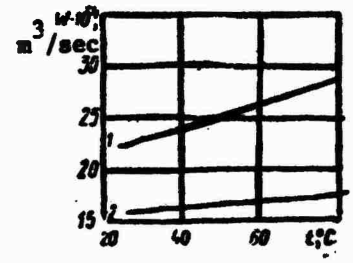

Figure 105. Oil flowrate as a function of oil grade and temperature: 1 - 75% MS-20 and 25% MK-8; 2 - 25% MS-20 and 75% MK-8.

Therefore statistical data are usually used to determine the required oil flowrate through the engine.

For preliminary calculations we can assume that the heat transfer into the oil for the single-contour TJE with the highest heat release rate amounts to 5.0-6.5 KJ/sec for every 1000 kgf of static thrust, while for the TPE the figure is 15-17 KJ/sec per 1000 KW of equivalent power. As a result of transfer of part of the heat to the air flowing in the outer flow, the heat transfer into the oil for the FJE is practically independent of thrust and for present designs lies in the range of 40-50 KJ/sec.

If we know the amount of heat which must be removed from the engine and the oil temperature difference, we can find the required oil flowrate through the engine.

Analysis of statistical data shows that for the TJE the required flowrate amounts to $(0.5\text{-}1.0)\cdot 10^{-4}$ m^3/sec per 1000 kgf of static thrust, and for the TPE the figure is $(4\text{-}6)\cdot 10^{-4}$ m^3/sec per 1000 KW of equivalent power. These flowrate values were found with an engine inlet oil temperature of 70-80°C. Deviations from these temperatures lead to variations of the oil flowrate through the engine. It is typical that these changes are more significant, the higher the viscosity of the oil used (Figure 105) or the steeper its viscosity-temperature curve.

To ensure reliable operation of the lubrication system, the pressure pump output is selected to be 1.5-2 times the required flowrate. The excess oil is bypassed by the pump relief valve into the suction line. As a result of this the required oil flowrate and the pump outlet pressure remain practically unchanged over a wide range of engine rotor speeds and aircraft flight altitudes (see Figures 91, 92).

Ceiling

The oil system ceiling is the maximal flight altitude to which the pressure pump provides the required oil flowrate throughout the engine. The oil system ceiling is considered to be adequate if with the maximal permissible oil temperature the oil pressure in the system does not fall below the minimal pressure at the service ceiling obtained using maximal engine power.

For powerplants which do not have means for regulating the oil temperature (oil systems with fuel-oil radiators), the system ceiling is determined for the maximal fuel temperature at the radiator exit permitted by the specifications.

In contrast with the fuel systems, in which at high altitudes the deterioration of the pump operating conditions is partially compensated by the reduction of the fuel flowrate, in the oil systems the required oil pressure and flowrate change very little up to the airplane ceiling. The basic problem in designing the oil system is to determine the oil pressure at the boost pump inlet, and not allow this pressure to decrease to values at which marked dropoff of the oil flowrate through the engine will occur.

By analogy with the fuel systems, the ceiling of the oil system can be found from the equation

$$p_H = p_{\text{вх min}} + \Delta p_r + \frac{V_м^2 \gamma_м}{2g} + \Delta p_и - y\gamma_м, \qquad (104)$$

where y is the height of the minimal oil level in the tank above the pressure pump.

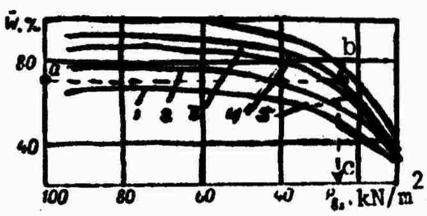

Figure 106. Cavitation characteristics of oil pump with air content in the oil: 1 - 30%; 2 - 20%; 3 - 10%; 4 - 5%; 5 - 0%.

The hydraulic resistance Δp_h is basically the frictional resistance. Therefore we attempt to make the suction line with smooth contours, without sharp bends.

Since the calculation of the oil system ceiling is made for steady level flight, the inertial pressure losses Δp_i can be taken as zero.

The minimal permissible pressure $p_{in\ min}$ at the pump inlet is found from the pump cavitation curves (Figure 106). After calculating the required oil flowrate through the engine from the condition of removal of the required amount of heat, we draw a horizontal line from this value on the ordinate axis. The value of $p_{in\ min}$ corresponding to the assumed percentage air content is obtained on the abscissa axis. Typical values of $p_{in\ min}$ for TJE pumps with good cavitation characteristics for the zero air content case amount to 60-80 mm Hg. Substituting these values into (104), we find p_H and then obtain the system ceiling from standard altitude tables.

In spite of the fact that the hydraulic resistance of the suction line is not large (20-50 mm Hg), all possible measures should be taken to further reduce this resistance, avoiding sharp changes of the line section area and flow direction. This is necessary for the following reasons. In the starting regime with cold oil the pressure in the system does not propagate uniformly in all directions. Therefore, for example, a larger pressure differential is required for oil flow through a bend or an elbow when the viscosity is higher. In the case of very high viscosity the bend or elbow may "block" the line, in spite of the fact that in the normal operating regime their resistance constitutes only a small part of the overall hydraulic resistance of the suction line.

If the oil system does not provide the required altitude performance, the pressure in the suction line is increased in order to maintain the minimal acceptable pressure at the inlet to the pressure pump up to the highest possible flight altitude. This is achieved by increasing the pressure in the tank or installing a boost pump in the suction line.

In contrast with the fuel tanks, for which pressurization from external sources is necessary, to increase the pressure in the oil tank we need only install in its vent line a constant differential pressure valve set to $\Delta p = 0.1\text{-}0.15$ kgf/cm^2. The differential pressure in the tank arises as a result of separation of air from the oil emulsion.

The boost pump installed in the suction line is connected in series with the main pressure pump. During climb to altitude the pressure at the inlet to the boost pump decreases and this leads to reduction of its output. The boost pump output is selected sufficiently large so that the pressure at the inlet to the main pump will not decrease up to the design altitude in this case. Part of the oil is bypassed by the boost pump relief valve into the suction line. As the pressure at the pump inlet decreases, the oil bypass through the valve decreases so that the pressure at the outlet from the boost pump remains constant. In this way it is possible to maintain a pressure at the pressure pump inlet which exceeds the minimal acceptable pressure.

Moreover, in order to increase the oil system ceiling it is necessary to reduce the hydraulic resistance of the suction line, use special oil system configurations with ejectors, increase the tank height above the pump, use pressurization systems, and install components and devices which provide better separation of the air from the oil.

Operation

Maintenance and servicing of the oil system are an integral part of the overall complex of operations in preparing the powerplant for flight and are carried out in the period when the preflight, postflight, and other forms of technical servicing are performed. During servicing of the lubrication system checks are made of the mounting of the tank, radiator, and other units. Attention is devoted to the absence of cracks, dents, and worn places in the lines. The oil quantity in the tank is checked and reserviced if necessary.

For each engine type only that grade of oil listed in the operations and maintenance manual is used. Arbitrary changing of the oil grade is not permitted, since this has a harmful effect on the operation of the friction surfaces and engine temperature regime and may lead to premature engine failure.

The check of the oil level in the tank permits early detection of problems arising in the engine or in the oil system. In all cases when the postflight check shows that no oil has been used from the tank or when the oil level in the tank has actually increased a check must be made for the presence of fuel in the oil. Causes for this phenomenon can be leaking in the fuel-oil radiator, leaks in the oil cavities of the fuel controller, or unintentional activation of the oil dilution system.

The quality of the oil in the system can be used to judge the condition of the friction surfaces, engine overheating which may have occurred, wear of the engine labyrinth seals, and so on. All these factors lead to change of the oil color and chemical composition, and of the foreign particle content in the oil.

Various techniques for monitoring the oil quality are being developed at the present time to provide earlier detection of engine problems. These include, for example, spectrographic and chemical analysis of periodically taken oil samples to establish the connection between the metallic particle content in the samples and initiation of deterioration of the friction surfaces. Magnetic plugs installed in various parts of the system are used. This method is quite simple; however, it permits the detection only of ferromagnetic particles, while the appearance of aluminum particles is characteristic for the beginning of the development of many engine problems.

Special signaling filters are installed on certain engine types in the centrifugal air separator oil cavity. When oil containing metallic particles passes through the filter sections, the slot-like gaps between the sections clog and this leads to closure of an electrical circuit and lighting of the "Oil Chip" warning light. The presence of chips can also be detected by means of sensors which react to change of the pressure drop across the oil filters.

The oil must be changed periodically to prevent excessive contamination. The basic methods used at the present time to determine the oil change intervals are test-stand and operational tests of oils in the engines. Subsequent modification of the oil service life is made on the basis of analysis of the operational experience of the given engine type.

The oil change intervals are listed in the maintenance manuals. The oil filters are flushed after the first test run and the first five hours of engine operation; thereafter they are flushed when performing periodic inspections for most engines. If metallic particles in the form of aluminum, bronze, or tin flakes are detected on the filters, an analysis must be made to determine the possible reasons for their appearance in the oil. To this end, investigations are made of the operations which have been performed on the engine to determine whether or not individual accessories, parts, or lines have been changed. After a careful inspection of the engine the oil filters are flushed, the oil in the system is changed, and the engine is started up and test run, after which the filters are checked again. If they are clean the engine is approved for further operation with subsequent inspection of the filters after the first trip. If it is found that the filter contamination is the result of failure of individual parts or components, further operation of the engine is not permitted. If an engine is changed because of the presence of metallic particles in the filters it is necessary to replace the oil radiator, flush out the tank, lines, and system components. The propeller is changed only if there are metallic particles on the governor oil filter.

Special attention must be paid to servicing the oil system after installing a new engine or changing the oil. In all cases the engine is started up and test run after the system is serviced; after the engine is shut down the oil level in the tank is checked and filled as necessary. For several engine types reservicing is performed after motoring the engine two or three times, followed by a check of the oil quantity in the tank after starting up the engine and making a test run.

The amount of oil to be serviced into the oil tank is indicated in the operating instructions or in the aircraft maintenance manual. Overfilling of the tank leads to oil loss through the vent system, and insufficient filling leads to engine overheating and decrease of the oil pressure in the system. Moreover, as a result of the reduced oil residence time in the tank, separation of the air from the oil is made more difficult and this lowers the system ceiling. Engine overheating may occur if the cooling surface of the air-oil radiators becomes contaminated.

During system maintenance a check must be made of the operation of the oil temperature regulators and the mechanisms for controlling the radiator flaps, since their improper operation leads to oil overheating or freezing in the radiator.

At low ambient temperatures it is recommended that the oil be diluted with gasoline in order to provide faster preparation of the engines for starting and facilitate the start. The use of diluted oil eliminates the necessity for draining the oil at low temperatures and preheating of the oil, engine, and oil system prior to starting. Moreover, as a result of the lower viscosity of the diluted oil the lubrication of the friction surfaces of the engine during starting is improved and the time required for engine warmup after starting is reduced.

The amount of gasoline required to dilute the oil depends on the initial oil viscosity, the assumed ambient temperature at the time of the next start, and the length of time the engine has been operated since the last previous dilution. Usually the ratio of the amount of gasoline to the amount of circulating oil amounts to 8-14%. During engine warmup the gasoline evaporates as the circulating oil temperature increases, and the engine begins to operate on oil with the original viscosity. The time in the course of which the gasoline is completely evaporated from the oil depends on the system capacity, the engine thermal regime, and the oil temperature.

Oil dilution in a system with a tank equipped with a circulation well is based on the dilution of the amount of oil in the engine, the system, and the well. The large quantity of oil located in the tank outside the well is not diluted. This shortens considerably the time for the gasoline to evaporate from the oil after engine startup (Figure 107). As the engine and the oil circulating through the well warm up, the remaining oil around the well also warms up and then gradually enters the well in place of the evaporating gasoline.

If after diluting the oil the engine is operated for less time than required for evaporation of the gasoline and then the engine is shut down for a considerable period of time, the amount of gasoline required to redilute the oil is determined from tables or special graphs.

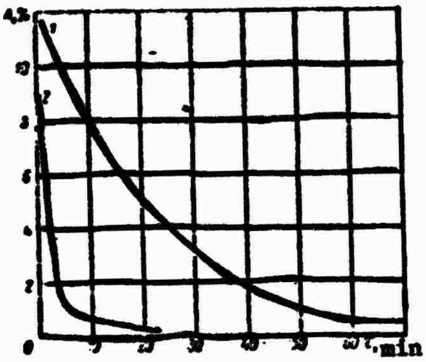

Figure 107. Gasoline quantity in oil as function of engine operating time: 1 - tank without circulation well; 2 - tank with circulation well.

At ambient air temperatures below 40°C, the oil must be drained from the system if the engine is to be shut down for an extended period (more than 2-3 hours). Servicing the system with hot oil (70-80°C) is accomplished immediately prior to engine start in preparing the aircraft for flight.

The oil in helicopter transmissions is diluted only once for the winter period. This is explained by the fact that the temperature of the transmission and the oil in the system is considerably lower than the engine temperature, and there is practically no evaporation of the gasoline from the transmission oil system.

For purposes of monitoring the condition of the oil in the transmission system, the maintenance manuals provide for a periodic check of the gasoline content in the oil on the basis of the specific gravity of the diluted oil. To make this check, 1-2 liters of oil heated to 20°C are drained from the system and its specific gravity is measured; then special tables or graphs are used to determine the percentile gasoline content in the oil.

The hypoid lubricants are diluted with AMG-10 oil. Usually the mixture composition consists of 2/3 hypoid oil and 1/3 AMG-10 oil. It is recommended that changeover to this mixture be made at ambient temperatures below 10°C. The diluted lubricants reduce the engine power required to overcome friction in the gearboxes, facilitate the operation of the friction clutch, and thus prevent premature clutch wear and failure.

CHAPTER 5

COOLING SYSTEMS

The powerplant must be maintained at a quite definite temperature to ensure its operating efficiency. Exceeding this temperature may lead to overheating of the engine, its accessories, and the airframe structural components located in the engine region. Overheating of the vitally important engine components (combustion chamber, turbine and so on) reduces their strength or causes failure. Therefore, the heat must either be removed from the hot parts of the engine into cooler parts or a coolant must be brought into contact with the hot parts. Cooling of the individual engine elements (for example, turbine blades and disks) is also accomplished with the objective of replacing the expensive high-alloy steels and alloys with other less expensive materials, reducing the thermal stresses by equalizing the temperature through the body of the part, and increasing the turbine inlet gas temperature.

The cooling may be accomplished in various ways, depending on the type of powerplant. For example, in the modern PE the cylinders are cooled by the oncoming airstream. Heat transfer from the hot parts of the engine can be accomplished by an intermediate medium (water, oil, fuel), which plays the role of coolant. In this case the heat is dissipated into the ambient medium through radiators. In addition to the powerplant, various units on the aircraft which themselves release heat must be cooled (generators, compressors, pumps, electrical transformers).

The need to create normal physiological conditions for the crew and passengers at high flight altitudes leads to the use of air bled from the gas turbine engine compressor for pressurizing the cabin. This is possible only if a special cooling system is provided to reduce the air temperature and maintain it constant in a given temperature range.

Increase of the flight speed increases the flight vehicle surface temperature. Structural heating reduces the material strength properties. The structural elements made from organic materials are the first to fail. A large temperature increase can lead to fuel boiling in the tanks and loss of the electrical insulating properties of rubber. In this connection the problem arises of selecting new materials which can operate at high temperatures. The use of aluminum and magnesium (without additional cooling) is limited to flight Mach numbers of about 2.2. The titanium alloys and stainless steels can be used at $M = 3$ with promise of further increase to $M = 3.5 - 4$.

At high flight speeds the powerplant is heated to higher temperatures than the outer surfaces of the aircraft, since the heat transferred from the engine and its equipment is added to the heat obtained from the decelerated flow. Under these conditions special cooling systems are required to maintain the temperature of the individual parts of the powerplant within the specified limits.

Thus, the powerplant cooling systems are designed to maintain the temperature of the individual accessories and systems within technically acceptable limits; ventilation of the cowled engine space in order to eliminate stagnant hot air zones and prevent accumulation in the nacelles of the vapors which are hazardous from the fire and explosion standpoint; protect the aircraft structural elements and its systems from engine thermal radiation.

The following requirements are imposed on the powerplant cooling systems: high efficiency, i.e., they must provide the required heat removal from the engine and its components (parts, assemblies, and units) with a small cooling flowrate and small power expenditure on operation of the cooling devices; simplicity of construction; minimal aerodynamic drag and low weight; reliable and simple, preferably automatic, control; operational simplicity (minimal time for preparation for flight, ease of inspection, installation, removal, and so on).

In addition to this list, the systems may be subject to other requirements associated directly with the specific type of aircraft and powerplant. The systems are verified during engine testing on the ground and in flight in the takeoff, climb, cruising, and landing regimes under conditions of maximal possible ambient air temperature (usually 45° C).

Classification

With regard to type of coolant used, the cooling systems are subdivided into two groups: gaseous (air) and liquid. In the first group use is made of atmospheric air or water vapor; in the second group use is made of fuel, oil, liquids with high boiling point, and molten metals. The coolant must have high values of the specific heat, heat transfer coefficient, and boiling point, must retain its physical properties during repeated heating and cooling cycles, must be safe, have low corrosive activity, and be simple to handle.

Depending on the coolant flow scheme, the systems may be open-circuit or closed-circuit. In the former the coolant is used once, after which it is released into the atmosphere. The closed system is characterized by the fact that the coolant circulates along a closed path, carrying heat from the hot parts to the heat exchanger.

Air cooling systems are used to protect compressors, generators, PE cylinders, combustion chambers, tailpipes, gas turbine engine disks and blades against overheating. These systems are the simplest in construction, do not require complex additional equipment, and are operationally reliable.

Heat removal from the hot parts by air is accomplished in two ways: either by blowing over the external surfaces of the parts (external cooling), or by circulating the coolant inside the parts (internal cooling). The first technique is usually used to reduce the temperature of assemblies located on the engine (compressors, pumps, generators) or of parts which are not directly wetted by the hot gas flow; the second technique is used for parts which are located in the hot gas stream.

Barrier cooling (Figure 108) has been widely used to reduce the temperature of the combustion chamber walls and turbine disks. In this case a jet of cold air is fed through orifices or slots in the direction of the hot gas stream and mixes with the hot gas to reduce its temperature and thereby protect the surface against overheating. A large pressure differential is not required to feed the cold air through the slot, and air taken from the atmosphere or the engine flow passage at the entrance to the combustion chamber is usually used. The drawbacks include the fact that in individual zones the cold air, having only a small excess pressure, may not overcome the hydraulic resistance of these zones, and therefore the appearance of local overheating is possible.

Figure 108. Different barrier cooling schemes:
1 - coolant; 2 - hot stream; 3 - wall.

For cooling of the PE, use is made of the approaching air stream, which flows over the finned walls of the cylinder heads and barrels. Increase of cooling effectiveness is achieved by placing the engine in a cowling (Figure 109). In addition, the cylinders are enclosed in baffles which squeeze the air stream against the cylinder finning and walls and thereby reduce the air mass flowrate. For better sealing, the outer edges of the baffles are trimmed with leather or special plastics. The reduction of the air mass flowrate through the cowling, achieved as a result of the presence of the baffles, reduces the required air passage exit area, which simplifies and improves the configuration of the cowling and the powerplant as a whole. The temperature of the individual parts is maintained in acceptable limits in different flight regimes by varying the air passage exit section area.

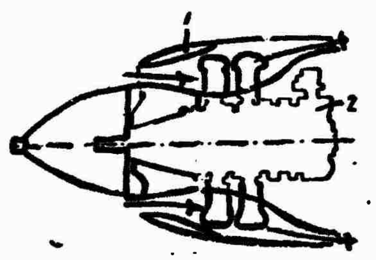

Figure 109. Piston engine installation in cowling: 1 - cowling; 2 - engine.

The specified thermal regime of the gas turbine engine powerplant is maintained by the air entering the cowled engine space through external intakes. These intakes are located on the nacelle (cowling cover panels), and their purpose is to supply the required amount of air for cooling the turbine case, tailpipe, compressor, generators, and other thermally loaded units.

For cooling of the turbine case and the tailpipe, it is recommended that the air be supplied by means of individual intakes, which should be installed aft of the firewall. The air leaves these zones at the trailing edge of the tailpipe. Air circulation in the tailpipe region in the absence of ram pressure is provided by ejectors. The simplest ejector is the exhaust nozzle ejector, in which a special shroud or the engine nacelle extends aft of the propulsive nozzle exit. To ensure satisfactory flow over the tail-pipe, it is sufficient for the length of the exhaust nozzle ejector to be 0.3 - 0.6 times the maximal diameter of the propulsive nozzle.

Equalization of the airflow velocity field in the cowled space in the tailpipe region is achieved in various ways. One, for example, is the installation of special baffles (frames) between the tailpipe shroud and the nacelle walls (Figure 110). Small openings are provided in the baffles to ensure air circulation between the shroud and the engine nacelle, which eliminates the formation of stagnant hot air zones. Increase of the effectiveness is also achieved by the use of screens, deflectors, and other devices to direct the air flow to the surfaces being cooled.

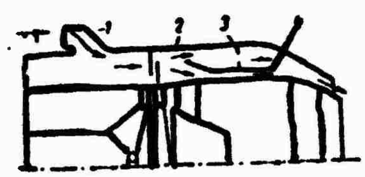

Figure 110. Cooling scheme using external air intakes:
1 - air intake; 2 - nacelle; 3 - baffle; 4 - shroud.

At high flight speeds the use of external air intakes is not desirable, since they increase the airplane aerodynamic drag significantly. Under these conditions various devices (valves) which provide for the entry of the required amount of air into the cowled space from the engine air intakes may be acceptable.

The systems for cooling helicopter powerplants, where forced air flow over the engines from a fan is required, deserve special attention. This requirement results from the fact that in the hovering and vertical takeoff regimes the engine operates under the most severe conditions and ram pressure is not sufficient for cooling

The air for cooling the PE helicopter powerplant (Figure 111) is taken through shutters installed at the inlet to the air duct, and after passing through the fan is directed to the engine and its

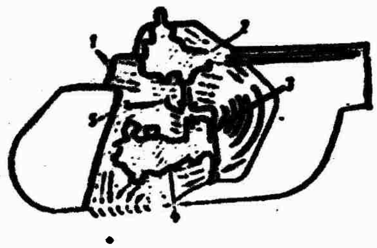

Figure 111. Cooling system of piston engine on helicopter:

1 - inlet shutters; 2 - transmission; 3 - fan; 4 - engine; 5 - warmup shutters.

accessories: transmission, generator, compressor, oil radiators, and so on. After this the air is discharged overboard through slots located on the lower and side panels of the fuselage skin. In order to reduce the cooling power losses the engine is enclosed in a special compartment, and its cylinders are equipped with baffles and guides which ensure that the cold air will reach the most highly heated parts.

Engine warmup during starting in the wintertime, and also prevention of overcooling in certain flight regimes, is achieved by controlling the shutters from the cockpit. When the shutters are closed the airflow from the outside is nearly blocked and the fan sucks in the warm air blowing over the engine cylinder fins through flaps which are opened specifically for this purpose. On helicopters with TPE the air is supplied from the fan through special ducts (air ducts) to the oil radiators, generators, pumps, and certain other components.

With increase of the flight speed, the cooling air temperature increases and the difference between the temperatures of the air and the hot body approaches zero. The limiting Mach number (Figure 112) to which air cooling is possible is found from the formula

$$M_{\text{пред}}=\sqrt{\frac{2}{r(k-1)}\left(\frac{T_{\text{гор}}}{T_H}-1\right)},$$

where r is the temperature recovery factor; for laminar flow in the boundary layer r = 0.85, for turbulent flow r = 0.88-0.89;

k is the adiabatic exponent.

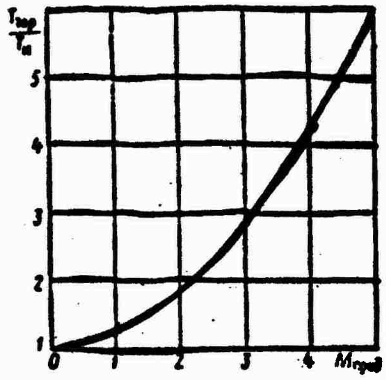

Figure 112. Ratio T_{hot}/T_H versus limiting flight Mach number.

At low supersonic flight speeds ($M \leq 2$) the air taken from the engine compressor has a temperature exceeding the airstream stagnation temperature. If the compressor air is directed to a heat exchanger which is exposed to the ambient airstream and then into an air turbine, its temperature can be reduced significantly. Systems with cooling turbines can be used along with gas turbine engine turbocompressors or special compressors.

In the cooling system shown in Figure 113, the hot air taken from the compressor is directed to a heat exchanger which is exposed to atmospheric air. Further reduction of the air temperature takes place in the evaporator, since heat is taken from the air by the evaporating water. Its temperature may be reduced to 100° C. The final cooling is accomplished in a turbine. A disadvantage of this system is the considerable water consumption, and the water supply must be on board the airplane.

The use of systems with cooling turbines requires the expenditure of a large amount of power. A large amount of air must be taken from the gas turbine engine for operation of the cooling turbine installation, and this reduces the engine thrust and increases the specific fuel consumption. Moreover, the heat exchangers have considerable aerodynamic drag.

Liquid cooling systems are being used at the present time in liquid rocket engines to reduce the combustion chamber wall temperature and in certain types of flight vehicles with TJE for cooling the oil in radiators. Fuel is widely used as the coolant. This

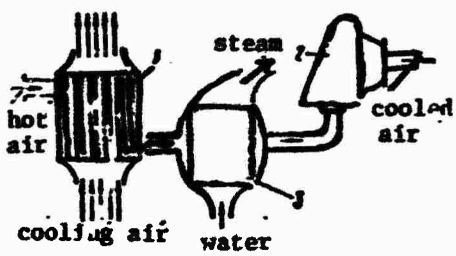

Figure 113. Cooling turbine installation with two-stage hot-air precooling:

1 - heat exchanger; 2 - turbine; 3 - evaporator.

technique can be used if there is the required temperature differential between the hot parts (combustion chamber wall) and the fuel, i.e., the condition $t_{fuel} < t^{*}_{hot \cdot body}$ must be satisfied. In this case the fuel temperature in the system must not exceed the maximal permissible temperature, defined from the viewpoint of fuel coking.

In the case of the high thermal fluxes in liquid rocket engine chambers, external cooling of the chambers becomes very difficult because of the large temperature difference between the inner and outer surfaces of the wall. If the fuel is used as coolant it may be found that its thermal capacity will not be sufficient to absorb the transmitted heat. Moreover, danger arises of overheating individual wall segments if local coolant boiling takes place. Under these conditions use is made of film cooling (Figure 114). A liquid is supplied through orifices or slots and forms a protective film on the surface. The film is carried by the gas stream along the surface and evaporates, absorbing heat in the process. The liquid vapors entering the boundary layer increase the layer thickness and reduce the heat transfer to the wall.

Film cooling may be supplemented by porous cooling (Figure 115), based on supplying a liquid (or gas) with either the same physical properties as those of the external flow or with different physical properties through a porous or perforated surface into the boundary layer. This technique requires less coolant consumption but its use involves several difficulties. One is the need to use a special porous material, which will have less strength than a solid material. Other drawbacks are the thrust loss when cooling the nozzles of reaction engines and the necessity to have the pressure differential

*Translator's Note: The original text mistakenly gives "cold" instead of "hot".

required to feed the coolant through the porous surface.

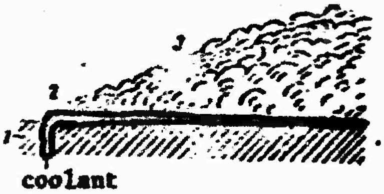

Figure 114. Film cooling:
1 - wall; 2 - liquid layer;
3 - vapor protective layer.

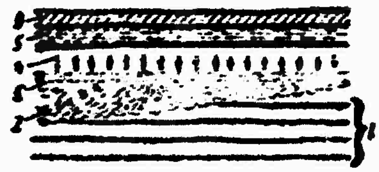

Figure 115. Porous cooling:
1 - combustion products stream;
2 - protective vapor layer;
3 - liquid film; 4 - porous wall;
5 - cooling liquid; 6 - outer
wall.

Techniques for Protecting Flight Vehicles and Their Powerplants Against Overheating at Supersonic Flight Speeds

The temperature distribution over the surface of a flight vehicle (Figure 116) depends on the flow conditions over the given surface, the flight time at supersonic speeds, the heat capacity and thermal conductivity of the materials.

The flight vehicle surface acquires a definite amount of heat from the boundary layer but scatters some fraction of this heat into space by radiation. Thus, only part of the thermal flux penetrates into the structure.

The heating of the flight vehicle structure at supersonic speeds takes place primarily from the outer surface. The material expands and elongates as it is heated. The structural elements located inside the flight vehicle have a lower temperature. The result is the appearance of large thermal stresses in the structure. In order to reduce the expansions of the individual elements relative to one another, use is made of a material with higher thermal expansion coefficient for the colder structure, sectional skins, and so on.

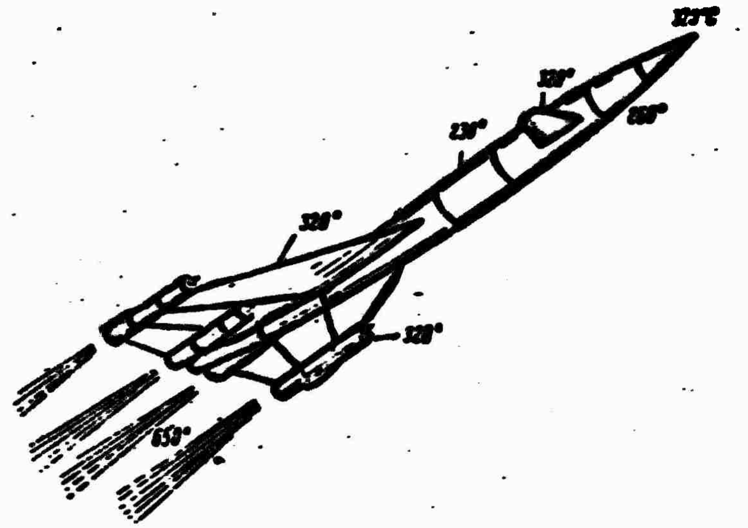

Figure 116. Supersonic airplane (M = 3) structural heating zones.

The simplest technique for protecting flight vehicles against overheating is the use of heat-resistant materials: steel, titanium alloys, beryllium, and so on. The windshields and side windows are made from temperature-resistant glasses, and thermal insulation coatings are used to protect the structure against heating.

At a particular flight speed the outer side of the wall will have the temperature T_{w1} (Figure 117), which is lower than the stagnation temperature. Since the wall offers definite thermal resistance to the heat flux passing through it, the temperature T_{w2} on the inner surface of the wall is somewhat less than that on the outer surface. Let us assume that the temperature T_{insl} at the outer side of the insulation is equal to the wall temperature T_{w1} in the absence of insulation. Since the insulation resists the thermal flux, the wall temperature under the insulation $T'_{w1} \ll T_{w2}$. Therefore the internal structural elements are heated to a lesser degree and the thermal stresses which develop in these elements are small.

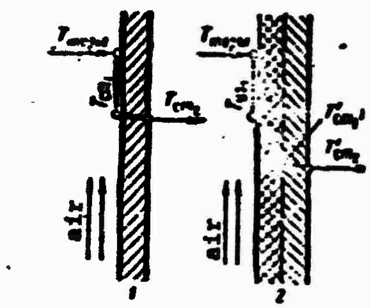

Figure 117. Heat transfer through wall:
1 - wall without insulation;
2 - wall with insulation

Another advantage of the thermal insulation coatings is that the high temperature of the coating outer layers results in decrease of the heat entering the coating from the boundary layer and considerable increase of the radiative heat scattering. It is advisable to install the thermal insulation between the outer and inner skins to reduce the heat transfer into the structure and retain a sufficiently smooth skin.

The material used for the insulation must have low thermal conductivity, high melting point, light weight, and good corrosion resistance. The porous and loose fibrous materials (asbestos, for example) have the lowest thermal conductivity. As the porosity and looseness increase,the thermal conductivity approaches that of air. However, the porous and loose fibrous materials have very poor mechanical properties and therefore can be used only for internal facing of the skin. The high-strength thermal insulation materials include some forms of plastics (silicones), fluoro-organic compounds, and the ceramic materials (aluminum oxide, for example). A drawback of these materials is their high thermal conductivity, and for the ceramic materials their brittleness as well.

The thermal stresses in the structure can be reduced by combined use of thermal insulation and cooling. In this case the cooling system is necessary to absorb the heat coming from the outer surface of the skin, and also to protect the internal flight vehicle compartments against overheating.

The relative weight of the flight vehicle protective system is the ratio of the weight of the flight vehicle skin with thermal protection to the skin weight without protection. We see from Figure 118 that for low steady-state stagnation temperatures all the systems are equivalent with regard to weight. The minimal structural weight is achieved by using a thickened uncooled skin (curve 2). With increase of the stagnation temperature the weight of the thermal protection increases. Thus, for temperatures above 600° C the minimal relative structural weight will be obtained with thermal insulation of the outer skin surface and cooling of the inner skin surface (curve 4).

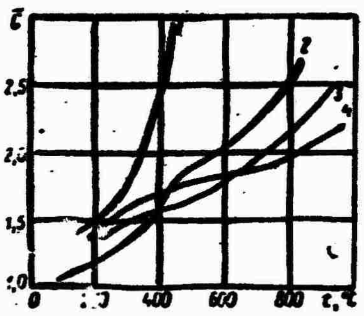

Figure 118. Relative weight of different thermal protection systems:

1 - cooling of inner skin, no thermal insulation; 2 - increased thickness of skin without insulation or cooling; 3 - evaporative water cooling; 4 - thermal insulation of outer skin surface and cooling of inner skin surface.

Among the other techniques for protecting the flight vehicle against overheating at high supersonic flight speeds, systems for cooling the outer surface with the aid of transpiration or mass transfer are used. Transpiration cooling is based on the principle of injecting water under a porous skin, vaporizing the water in the skin layer, with subsequent discharge of the vapor at the skin surface, which alters the nature of the velocities and temperatures in the boundary layer and reduces the heat transfer and temperature recovery coefficients markedly. The advantage of transpiration cooling is that it provides direct cooling of the outer skin surface.

To provide mass transfer cooling the surfaces are coated with subliming or dissociating materials (gypsum, magnesium nitride, graphite). The use of this sort of coating makes it possible to avoid the complications associated with the use of the porous skin and lighten the structure, since there is no need for additional equipment (coolant tanks, pumps, lines) or cooling system controls.

Analysis

Radiators

Check calculations of radiators are made if the radiator geometric characteristics (cooling surface area, number of tubes, their length, diameter, and so on) are known and we are required to determine the amount of heat transferred from one heat transfer agent to the other, and also the final temperature of the heat transfer agents (working fluids). Design calculations are made in the development of new radiators to determine the cooling (heat exchange) surface area.

We shall consider as an example the technique for making the check calculations in application to fuel-oil radiators, i.e., from several radiator types we select that one which provides removal of the required amount of heat from the engine. In addition to the radiator geometric characteristics and the required amount of heat which must be removed, we need to know the values of the fuel and oil flowrates W_f and W_0 through the radiator, the radiator inlet temperatures $t_{f\ in}$ and $t_{0\ in}$, and the physical properties of the fuel and oil (viscosity, density, specific heat, thermal conductivity, and so on).

The amount of heat which the radiator must remove is

$$Q_{\text{расп}} = KS\Delta\bar{t} \text{ Jm/sec} \tag{105}$$

where K is the heat transfer coefficient, $J/m^2 \cdot sec \cdot deg$;

S is the cooling surface area, m^2;

$\Delta\bar{t} = \bar{t}_0 - \bar{t}_f$ is the average temperature difference.

Taking the temperature change along the length of the heat exchange surface to be linear (which is valid for small variation of the temperature along the surface), we can write

$$\bar{t}_{\text{м}} = t_{\text{м.вх}} - \frac{\Delta t_{\text{м}}}{2}; \quad \bar{t}_{\text{т}} = t_{\text{т.вх}} + \frac{\Delta t_{\text{т}}}{2},$$

where $\Delta t_f = t_{f.\ out} - t_{f.\ in} = Q_{req}/c_f W_f \rho_f$ is the fuel temperature rise in the radiator;

$\Delta t_o = t_{o.in} - t_{o.out} = Q_{req}/c_o M_o \rho_o$ is the oil temperature drop in the radiator (here and hereafter in the text the subscript f applies to the fuel and o to the oil).

If the value of the heat transfer coefficient is not known it can be found from the formula

$$K = \frac{1}{\frac{1}{\alpha_{\text{т}}} + \frac{1}{\alpha_{\text{м}}}}.$$

The heat transfer coefficients α_f and α_o are found from the relation

$$\alpha = \frac{\mathrm{Nu}\,\lambda}{d},$$

where Nu is the Nusselt number, which characterizes the convective heat transfer between the liquid medium and the heat exchange surface;

λ is the thermal conductivity (of the oil, fuel), J/m·sec·deg;

d is the tube diameter (outer in calculating α_o and inner in calculating α_f), m.

We know from thermal process similarity theory that the Nusselt number is a function of two other numbers: the Prandtl number $\mathrm{Pr} = c_p \rho \nu / \lambda$ and the Reynolds number $\mathrm{RE} = Vd/\nu$, in which c_p is the specific heat of the fluid at constant pressure (c_o and c_f in the present case), V is the fuel (oil) velocity in the radiator.

Since V = W/F, to determine the fuel (oil) velocity in the radiator we need to know the flow section areas. For the fuel, which flows throught the tubes, the flow section is

$$F_{\text{т}}=\frac{\pi d^2}{4}\cdot\frac{n}{z_{\text{т}}},$$

where n is the number of tubes in the radiator;

z_f is the number of sections in the radiator fuel cavity.

For the oil, which travels between the tubes,

$$F_{\text{м}}=(x+1)h_1 l;$$

where x is the number of tubes in the narrowest cross section of the bundle;

h_1 is the distance between tubes (Figure 119);

l is the tube length.

In calculating the Reynolds and Prandtl numbers the parameters appearing in the formulas should be taken for the average values of the fuel and oil temperature in the radiator. The further calculations amount to the following. For laminar fuel flow in the tubes ($Re \leq 2\cdot 10^3$) $Nu = 0.17\ Re^{0.33}\ Pr^{0.43}$. In the transition regime ($2\cdot 10^3 \leq Re \leq 10^4$) $Nu = k_0\ Pr^{0.43}$ (Figure 120). For strongly developed turbulent flow ($10^4 \leq Re \leq 5\cdot 10^6$) $Nu = 0.021\ Re^{0.8} Pr^{0.43}$. This last formula can also be used to calculate α_0 for longitudinal oil flow around the tubes.

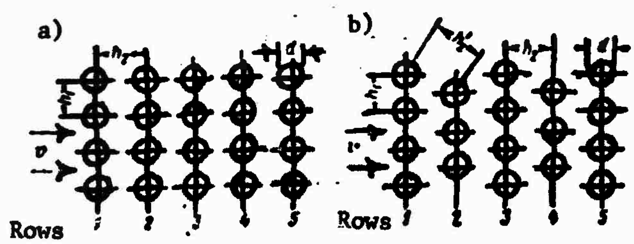

Figure 119. Tube arrangement in bundle:
a - corridor arrangement; b - staggered arrangement (h_1, h_2, h'_2 are the transverse, longitudinal and diagonal bundle pitch).

For transverse oil flow around the tubes we must take into account the tube placement in the bundle (see Figure 119). If the tubes are staggered, which is most typical of oil radiators, then $Nu = 0.41\, Re^{0.6} Pr^{0.33}$, for the in-line arrangement $Nu = 0.23\, Re^{0.65} Pr^{0.33}$. From the known values of Nu_0 and Nu_f we find the heat transfer coefficients α_o and α_f and then the heat transfer coefficient K.

We see from Figure 121 that with increase of α_0 (taking α_f to be constant) a relatively rapid increase of K takes place until α_0 and α_f become comparable. With further increase of α_0 the increase of K slows down and then practically stops. From this we can draw the following conclusions. For $\alpha_o = \alpha_f$ the heat transfer can be intensified (increase of the heat removal per unit radiator surface area) by simultaneous increase of α_o and α_f. If $\alpha_o \neq \alpha_f$, intensification can be achieved only by increasing the smaller of the two coefficients. Having K, S and $\Delta\bar{t}$, we use (105) to find Q_{avail}.

Making the assumption that in the operating temperature range the physical properties of the fuel and oil remain constant, we can plot nomograms which permit finding Q_{avail}/S graphically and then Q_{avail} (the nomogram of Figure 122 is calculated for T-1 fuel and MK-8 oil).

We compare the resulting Q_{avail} with Q_{req}. If the difference between them does not exceed 10% the radiator has been selected properly. If the difference is more than 10% a recalculation must be made, i.e., we select a radiator with different cooling surface area. Upon completing the calculations, we verify that the fuel temperature at the radiator outlet does not exceed the maximal permissible value, i.e.,

$$Q_{расп} = \alpha_в S \Delta t$$

The heat transfer Equation (105) can be used in the design calculation to determine the cooling surface area S. The amount of

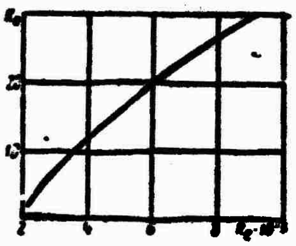

Figure 120. k_0 versus Re.

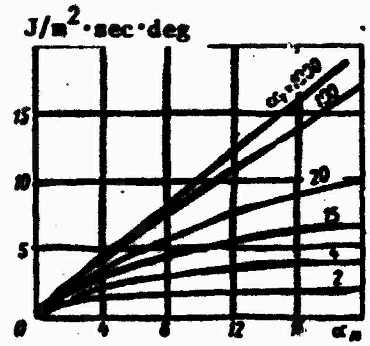

Figure 121. Curves of $K = f(\alpha_0 \cdot \alpha_r)$.

heat to be removed from the engine must be known in this case, and the problem reduces to calculating the heat transfer coefficient and then the cooling surface area.

In selecting the cooling surface area of the air-oil radiators, we must remember that a given amount of heat Q_{avail} can be dissipated into the ambient medium for different values of S by varying the flow velocity over the radiator. In fact, for turbulent air flow in the tubes $Nu = 0.018\ Re^{0.8}$, then $\alpha_a = C(\gamma V_p)^{0.8}$. The coefficient $C = \frac{0.018\lambda}{d^{0.2}(\mu g)^{0.8}}$ is found from the average values of the air temperature in the radiator (μ is the dynamic viscosity in kgf·sec/m^2). Since $Q_{avail} = \alpha_a S \Delta \bar{t}$ (we assume that $K \approx \alpha_a$), for given $\bar{\Delta t}$ the product $\alpha_a S$ is a constant. Increase of the flow velocity over the radiator leads to increase of the heat transfer coefficient, reduction of the radiator cooling surface area (Figure 123) and weight (Figure 124).

Decrease of the cooling surface area leads to a corresponding decrease of the required radiator frontal area F_r (see curve 2 in Figure 124), since the ratio S/F_r is a constant and for most current designs amounts to 120-130. At the same time, increase of the air velocity ahead of the radiator increases the power losses in overcoming the drag of the radiator installation. Therefore, in making the calculations we need to find that velocity V_r for which the overall power expenditures will be minimal. Usually we assume that the air velocity ahead of the radiator

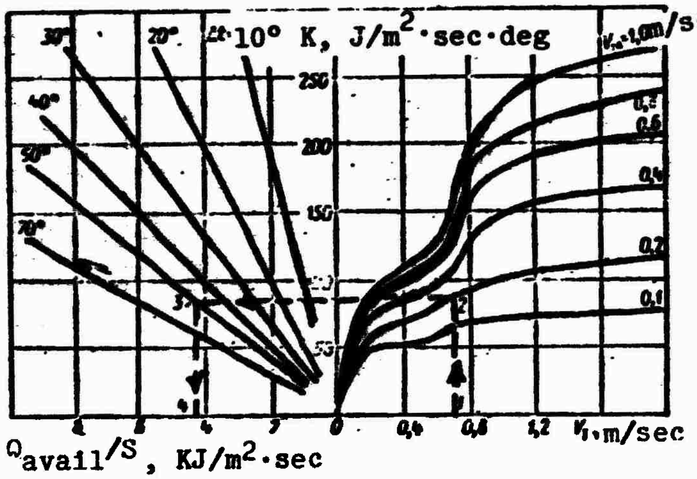

Figure 122. Nomogram for determining Q_{avail}.

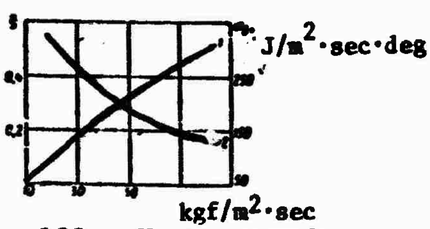

Figure 123. Heat transfer coefficient (curve 1) and surface cooling coefficient (curve 2) versus specific air mass flowrate through radiator.

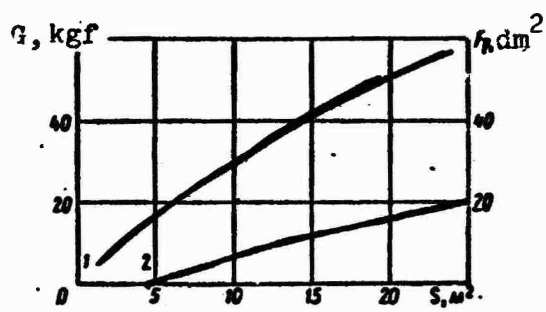

Figure 124. Weight (curve 1) and frontal area (curve 2) versus radiator cooling surface area.

$$V_{p}=aV_{обд},$$

where a is the air flow coefficient (the optimal value of a = 0.1-0.15);

$V_{cool} = V_0\sqrt{1+B}$ is the flow velocity over the radiator with account for propeller operation (Figure 125);

V_0 is the airstream velocity;

$B = \frac{P}{F_{disk}\frac{\rho V_0^2}{2}}$ is the propeller disk loading;

P is the propeller thrust.

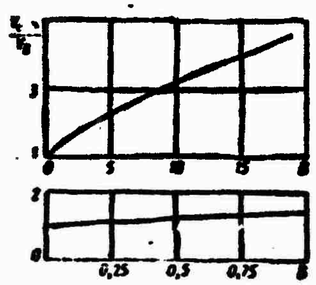

Figure 125. V_{cool}/V_0 versus disk loading $/B$.

For the known values of V_r, after determining the quantity $(\gamma V_r)^{0.8}$ for the design case, we find α_a and then the cooling surface area. The air velocity in the radiator tubes is

$$V_{\tau p}=\frac{V_p}{f_{ж}},$$

where f_{ar} is the area ratio, i.e., the ratio of the total air passage area to the radiator frontal area; for oil radiators $f_{ar} = 0.45 - 0.55$.

Upon completion of the thermal calculations a hydraulic analysis is made of the radiators, i.e., we determine the magnitude of the coolant pressure loss as it flows through the radiator.

Tailpipes

Analysis of tailpipe cooling involves determining the air quantity necessary to maintain an acceptable tailpipe wall temperature during operation. We represent the schematic of the cooling system in the following form (Figure 126): the heat transmitted from the hot gases to the wall is removed from the wall by convective heat exchange with the air flowing along the duct between the engine nacelle (shroud) and the tailpipe. To ensure normal cooling conditions the required air flowrate is

$$W_в=\frac{Q}{c_p \rho_в \Delta t}\ м^3/сек,$$

where ρ_a is the air density, kg/m^3;

Δt is the permissible air temperature rise in the cooling duct; usually $\Delta t = 50 - 80°$ C.

The amount of heat transferred from the wall to the air is

$$Q=\alpha_в S(t_{ст}-\bar{t}_в),$$

where $\alpha_a = f(\gamma, V_{cool})$ is the coefficient of heat transfer from the wall to the air. It is determined using the technique described above (see analysis of radiators). For approximate calculations we can take α_a = 200-300 J/m^2·sec·deg;

S is the wall surface area, m^2;

t_w is the wall temperature, deg;

$\bar{t}_a$ is the average air temperature in the annular duct, deg.

After determining the required air flowrate, we can calculate the annular duct area

$$F = \frac{W_в}{V_{охл}} \text{ м}^2.$$

Analysis of statistical data shows that the airflow required for cooling the TJE amounts to 1.5 - 2% of the total air flowrate through the engine.

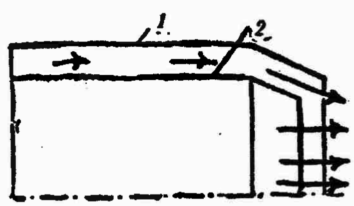

Figure 126. Tailpipe cooling scheme:
1 - shroud; 2 - tailpipe.

In order to obtain the required cooling flow velocities V_{cool} with small air flowrate around the tailpipe, it is recommended that special shrouds be installed which reduce the area of the cross section through which the air flows. Moreover, the shroud protects the engine nacelle walls from heating by thermal radiation.

In analyzing cooling systems, we can also solve the inverse problem, i.e., we can find the required cooling flow velocity for a given amount of heat removed. Since

$$\alpha_в = \frac{Q}{S(t_{ст} - \bar{t}_в)},$$

after determining α_a we use dimensionless relations to find the

Nusselt number and then the Reynolds number and from it the cooling flow velocity. For the most highly thermally loaded tailpipe walls the cooling flow velocity $V_{cool} = 30 - 50$ m/sec.

The cooling system analysis is made for the maximal flight altitude, i.e., for the altitude of minimal cooling air mass velocity. At altitudes lower than the design altitude V_{cool} is greater than the required value, which leads to increase of the cooling system drag. Sometimes an adjustable cooling duct discharge area is used to eliminate this problem.

Maintenance

The condition of the cooling systems is checked when performing the operations on the powerplant specified by the maintenance instructions. Inspections are made of the cowling and engine nacelle skin; absence of deformations, cracks, dents, and loose rivets is verified. A check is made of the condition of the cowling panel locks and the condition of the profiled rubber seals, condition of the tailpipe, paying special attention to the welded joints. A check is made of the clearances between the tailpipe and engine nacelle, since deviation from the acceptable clearances between these surfaces leads to local overheating of the tailpipe or nacelle walls. If traces of overheating are found on the gas passage structural components, the reason for these indications is investigated and repairs are made if necessary. During engine removal a careful examination is made of the condition of the nacelle structure (frames, stringers, fittings attaching nacelle to the wing), checking for cracks, corrosion, rivet failures, and loose bolts. A check is made of the condition of the rubber seals at the ducting joints, air intakes, and firewalls. Inadequate sealing of the air ducts causes air leakage through the slots and results in increase of the flight vehicle drag. Cleanliness of the cooling surfaces is verified, since dust and dirt reduce the heat transfer.

In turboprop powerplants a check is made of the condition of the air-oil radiators, security of their mounting, cleanliness and integrity of the honeycombs. Checks are made of the operation of the mechanisms for controlling the radiator flaps and the PE cowling exit flaps, checks are made for unacceptable play in the flap support fittings and control system, and a check is made that the flaps fit tight against the duct skin. Poor mating of the flaps can lead to overcooling of the engine or individual components of the engine during extended flight vehicle gliding. Excessive flap opening causes flow separation, which reduces the air flowrate and increases the cooling system drag.

During operation of helicopter cooling systems particular attention is devoted to the fan, checking for cracks, dents, nicks, and other mechanical damage to its blades. Provisions are made for tightening the nuts attaching the fan to the engine shaft when performing periodic inspections, and also periodic removal of the fan to verify the condition of the splines and threads on the engine shaft. Loose fit of the impeller hub on the engine shaft leads to impeller wobble, powerplant vibration, and the appearance of fatigue cracks on the hub. It is recommended that the shutters which provide air entry into the fan compartment be closed when the helicopter is parked in order to prevent foreign object entry into the fan compartment.

CHAPTER 6

ANTI-ICING SYSTEMS

Icing Conditions

Modern aircraft fly under the most varied climatic conditions and over routes with strict adherence to cruising altitudes. Therefore it is practically impossible to avoid encountering regions of space in which icing may occur. Powerplant icing is affected by numerous factors, the primary ones being the temperature and relative humidity of the air, cloud water content, mean droplet diameter, aircraft flight speed and altitude.

The meteorological conditions which favor icing are characterized by the presence of supercooled water droplets or ice crystals which are suspended in the air in the form of clouds, fog, rain, wet snow, and so on.

The aircraft disturbs the supercooled water droplets present in the air from their stable equilibrium and the droplets freeze on the aircraft surfaces.

Thus, powerplant icing is the process of forming ice deposits on the surfaces of the engine intakes or other parts of the powerplant during the time the aircraft is flying in meteorological conditions which favor ice formation.

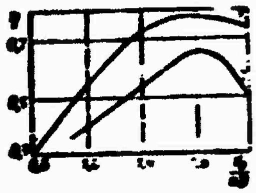

Figure 127. Propeller efficiency variation: 1 - clean; 2 - iced.

During icing of gas turbine engines, ice is formed on the stationary surfaces of the air inlet duct, guide vanes, engine nose fairing, first compressor stage blades, leading edge of the inlet diffuser (nose cowling), intakes for the generator cooling air, and other air intakes located on the engine cowling.

We differentiate the following types of ice formation:

clear ice in the form of a glassy film with smooth surfaces. It forms during flights in supercooled rain or drizzle. Deposition of such ice is most often observed in the ambient temperature range ± 5°C;

rime ice, which is opaque ice of rough, grainy, or crystalline structure. Such ice forms during flights in clouds containing a large number of supercooled droplets of different size in the temperature range from 0 to minus 10°C;

crystalline ice, which deposits in small quantities at temperatures below minus 10°C and during flights in clouds consisting of very fine supercooled water droplets. Sometimes this form of ice is termed frost. Mixed ice formation types can also be observed. The icy deposits of irregular form which are formed during flight in areas where rain and snow are falling present the greatest hazard.

In contrast with the airframes, for which icing occurs at negative temperatures, the gas turbine engines may be subject to icing at ambient temperatures up to 5-10°C. During engine operation on the ground or at low flight speeds, the air is sucked into the inlet duct and expanded, as a result of which the air temperature decreases and may reach values at which icing can take place. In powerplants with PE and TPE propeller icing starts first, beginning with the hub or spinner.

Figure 128. Icing of axial compressor blades obtained with engine operating on test stand under conditions of high water content.

Figure 129. Blade tip bending resulting from entry of pieces of ice into the compressor.

Propeller icing presents a severe flight hazard since there is a marked reduction of the propeller efficiency (Figure 127), and also propeller unbalance develops and leads to severe engine vibration. Pieces of ice which break loose from the propeller will damage the aircraft skin.

Ice formation on the surface of the inlet duct and directly on the entrance to the compressor reduces the air flowrate and engine thrust. If the pressure ratio across the propulsive nozzle is subcritical, reduction of the engine inlet cross section leads to increase of the fuel flowrate to maintain a given engine rotor speed. As a result of this the turbine inlet temperature increases, the operating condition approaches the boundary of the region of unstable compressor operation, and this causes engine vibration and possible engine flameout. Moreover, the gas temperature increase creates a danger of overheating the nozzle vanes and turbine blades. Gas turbine engines with axial compressors are particularly sensitive to icing. In these engines intense ice formation takes place on the inlet guide vanes and also on the first row of compressor rotor and stator blades (Figure 128). The ice may break loose and cause compressor damage (Figure 129).

low weight and small size;

quick response time;

it is desirable that automatic activation and deactivation be provided from special icing initiation sensors;

simplicity of maintenance and operation; possibility of checking system condition on the ground and in flight.

Classification

The thermal systems are most widely used to protect powerplants against icing. Depending on the energy sources, they are subdivided into the hot-air and electro-thermal types. The former use the thermal energy of air taken from the engine compressor. The higher the temperature and pressure of the air downstream of the compressor, the more effectively these systems operate. When the air flowrate through the engine is small (PE,TPE), hot air may be obtained by means of heat exchangers heated by the exhaust gases.

Antiicing systems may be either continuous duty or cyclic. The continuous systems do not permit ice formation on the protected surfaces. They are used to protect against ice formation on the engine air intakes and parts located in the air intake duct, for which ice formation and subsequent removal may disrupt normal engine operation or cause engine damage. The cyclic systems periodically remove the ice layer which forms on the protected surfaces.

A typical continuous-duty hot-air anti-icing system is shown in Figure 130. This system supplies hot air from the compressor to an annular chamber located at a depth of 200-250 mm in the air intake leading edge. The chamber is separated from the engine cowled space by a sealed partition. When the valve 6 is opened hot air enters the chamber 1. After passing through the annular duct, the air is discharged into the cowled space. The hot air supply to the annular chamber may be provided at several points along the circumference in order to create a uniform temperature pattern. Reduction of the temperature and increase of the quantity of the entering air are

During powerplant icing the ice forms on the entrance to the air intakes for cooling the generators, oil radiators, compressors, tailpipes, and other equipment, with the result being deterioration of the cooling conditions of these devices with subsequent overheating and failure.

In order to ensure normal engine operation under icing conditions, it is necessary to develop special protective equipment, i.e., anti-icing systems. The operating principle of most such systems is based on the fact that their activation raises the temperature of the protected surfaces above freezing and ice formation on these surfaces becomes impossible.

Requirements

The requirements imposed on the anti-icing systems include the following:

ensure flight safety under all icing conditions and in all engine operating regimes;

reliability and efficiency of operation under different meteorological conditions over a wide range of flight speeds and altitudes, unlimited operating time;

possibility of regulation of heating intensity as a function of ambient air temperature and icing intensity;

safety of system operation when parked and taxiing, high rate of heating of the protected surfaces;

engine air intakes and all parts which protrude into their ducts must have a continuous-duty anti-icing system to prevent ice formation on the protected surfaces both on the ground and in flight;

installation of icing initiation warning devices and icing intensity indicators, minimal triggering time of the sensors, high sensitivity and absence of false triggering;

fire safety;

minimal energy consumption;

absence of interference with operation of radio communications and navigation equipment;

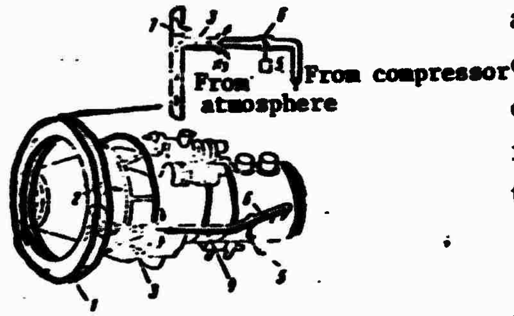

Figure 130. Air intake anti-icing system: 1 - antiicing chamber; 2 - air exit flange; 3 - ejector; 4 - line; 5 - electric actuator; 6 - valve; 7 - baffle.

accomplished by an ejector, which draws air from the space inside the cowling, mixes it with the air coming from the compressor, and supplies the mixture to the heated cavity.

Heating of the air intake leading edge can be accomplished using the scheme shown in Figure 131. Here the hot air enters the annular chamber A, from which it flows along slotted ducts 1 into the cavity B, heating the air intake leading edge skin 2. The air is discharged into the atmosphere through the line 3, located in the lower part of the air intake.

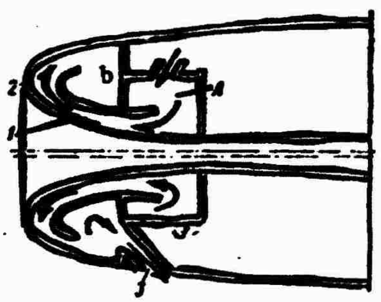

Figure 131. Air intake leading edge heating scheme.

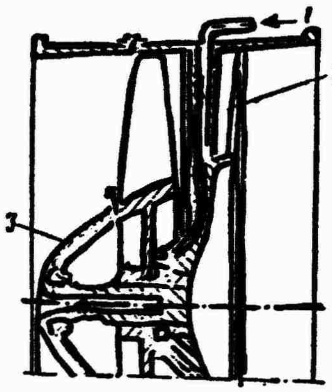

Figure 132. Engine inlet heating scheme.

Air taken from the engine compressor is used for anti-icing of the structural elements located in the engine inlet passage (spinner, inlet guide vanes, air distributing baffles, and so on) (Figure 132). In this scheme the hot air flows through the line 1 into the guide vanes 2 (a schematic of the air circulation in the vanes is shown in Figure 133), heats the vanes, and then flows to the leading edge of the spinner 3. Then the air flows through an annular channel formed

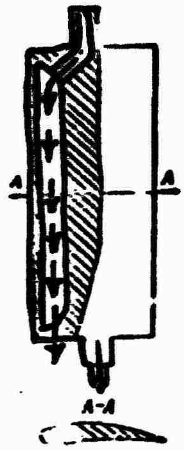

Figure 133. Air circulation in leading edge of guide vane.

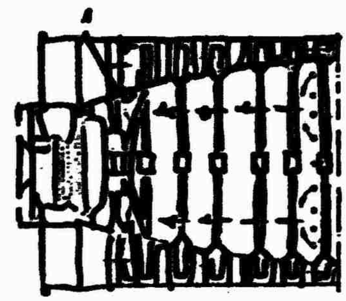

Figure 134. Engine inlet guide vane heating scheme.

by the outer wall and an inner baffle along the spinner and is discharged through openings into the engine inlet duct.

Heating of the inlet guide vanes can also be accomplished as shown in Figure 134. Here the air is directed along special drilled passages from behind one of the compressor stages into the internal engine rotor cavity and then through ports in the disk diaphragms into the cavity A. From this cavity the air flows to the vane roots, along an internal passage in the body of the vane, and then exits into the engine inlet air passage through openings in the upper part of the vane.

A disadvantage of the hot-air systems is their low effectiveness when the engines are operating at conditions near idle because of reduction of the air flowrate and temperature. This applies particularly to engines with a low pressure ratio. Moreover, when the hot-air system is activated there is some reduction of the power (thrust) and increase of the engine temperature conditions, and increase of the specific fuel consumption. For the TJE the thrust reduction is nearly directly proportional to the amount of air bled from the engine. The TPE are even more sensitive to air bleed. Every percent of air bled by the anti-icing system leads to a 1.5-2% reduction of the TPE power and a 1-1.5% increase of the specific fuel consumption. The maximal amount of air which can be bled from the engines should not exceed 12% for the TJE, 7% for the FJE, and 5% for the TPE. Failure to observe these conditions and poor choice of the air bleed locations can lead to engine overheating, disruption of the velocity pattern uniformity downstream of the compressor, blade vibration, and increase of the gas temperature pattern non-uniformity at the turbine inlet.

Use of the hot-air system is advisable for air intakes with leading edge radii of curvature greater than 6-8 mm.

These systems become ineffective for the sharp supersonic air intake leading edges, and this leads to the necessity to use the electro-thermal systems.

The electro-thermal system does not have the disadvantages inherent in the hot-air system. In spite of this, they were not used for a long time because of the necessity for highpower electrical power sources. This system consists of heating elements, a programmer, power sources, contactors, and electric wiring.

The heating element is usually made in the form of thin metallic foil which can be applied in sections, current conducting paints, strip heaters fabricated by deposition and insulated by an epoxy resin, wires, or in the form of a current-conducting rubber layer. These elements, located between insulating layers, are termed heating packets. After attachment to the surface to be protected against icing, the packet is covered externally by an overlay to provide protection against mechanical damage and erosion.

The construction of the heating packet makes it possible to obtain the most favorable energy distribution over the profile being heated, and also makes it possible by altering the insulation layer thickness to direct the heat flux in the direction required for heating. The electro-thermal system has a higher thermal utilization factor. The energy available is practically independent of the engine operating regime, flight altitude and speed, and ambient air temperature. Moreover, the system can be installed on powerplant parts of any size and shape. These systems are used to protect propellers and supersonic air intakes against icing.

They are fed from dc or ac (preferable three-phase) generators. The dc system is better from the viewpoint of construction simplicity; however it is considerably heavier than the ac system of equivalent thermal output.

The electro-thermal systems may be continuous-duty or periodically (cyclically) heated. The primary objective of the continuous heating systems is to prevent icing of the protected surfaces. The cyclic systems permit ice to form; however the operating cycle of these systems must be selected so that ice deposits of hazardous dimensions do not form on the protected surfaces. A second mandatory condition for the use of such systems is the possibility of complete removal of the ice in a single cycle. The continuous heating systems are usually used to protect units requiring relatively little power, or where ice formation cannot be permitted on the protected surfaces because of their operating conditions (for example, the engine air intakes).

The essence of the cyclic heating system is that the entire surface being protected against icing is divided into several sections consisting of heating packets. Each section is connected for a short time to the energy sources by means of a special device (programmer), and then heating of this section terminates. Thus, the entire usable power is periodically, in a definite sequence, concentrated on the individual segments of the heating surface. This type of heating yields considerable savings in electrical energy. This is a result of the fact that in the cooling regime a thin layer of ice forms on the protected surface, and this reduces markedly the heat removal from the surface by the approaching airstream.

In contrast with continuous-duty heating, in cyclic heating the energy is not expended on melting the entire ice mass which has deposited on the protected surface, but rather on melting only the small part of this mass which is necessary to destory the bonding of the deposited ice with the protected surface. Thereafter the melting ice is carried away by the approaching airstream. The ice must break loose quite rapidly. This condition must particularly be satisfied for helicopter lifting rotors, since ice formation on the blade surfaces leads to excessive increase of blade drag, reduction of the lifting qualities, reduction of the PE crankshaft speed, and "driving" of the helicopter control stick. Moreover, asymmetric removal of the ice leads to main rotor unbalancing and increasing vibrations. For fast and complete removal of the ice from the blades, it is necessary to use very large specific powers for the shortest possible heating time. To provide simultaneous removal of the ice from the entire blade length it is desirable that the specific heating power vary from a maximal value at the root to the minimal value at the blade tip (Figure 135).

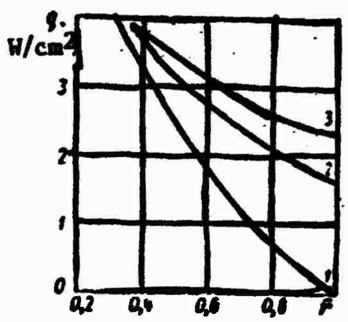

Figure 135. Variation of specific heating power along blade for ambient temperatures: 1 - minus 10°C; 2 - minus 20°C; 3 - minus 30°C.

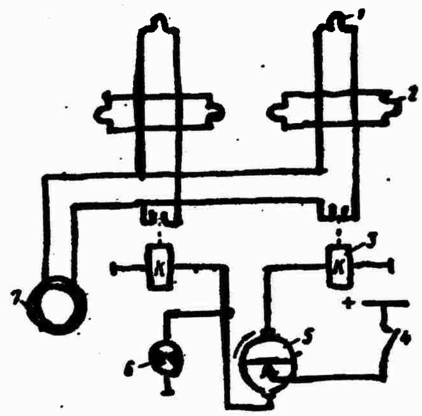

Figure 136. Propeller anti-icing system: 1 - propeller spinner heating element; 2 - blade heating packet; 3 - contactor; 4 - on-off switch; 5 - programmer; 6 - warning light; 7 - generator.

Let us consider as an example the operation of an electro-thermal air propeller anti-icing system (Figure 136). The power supply sources for this system are ac generators 7 (115 V, 400 Hz). The controlling contactors are activated by pulses transmitted by the programmer 5. The latter ensure operation of the propeller anti-icers in accordance with the prescribed cycle. When the contactors are closed, voltage from the generators is supplied through brushes and slip rings to the heating packets 1 and 2, located on the blade leading edges and propeller spinners. The operation of the anti-icing system is monitored by illumination of a signal light connected to one of the sections or by the indications of ammeters installed in the generator circuit.

Because of its small profile dimensions, the helicopter tail rotor is more sensitive to icing than the main rotor. Therefore it is advisable to equip the tail rotor with a continuous heating system or reduce the cycle time by a factor of three or four. Some Soviet helicopters are equipped at the present time with a cyclic heating system with controllable (as a function of the icing conditions) cycle time. Thus, on the Mi-6 helicopter activation of main and tail rotor blade heating is accomplished by a selector switch which permits the use of one of the following regimes, depending on the icing conditions: heating for 20, 40, 60 sec and cooling for 100, 80, 60 sec, respectively.

In addition to the systems mentioned above, the following thermal systems can be used to protect powerplants against icing:

oil heating of the protected surfaces. This technique is analogous to hot-air heating; however it is considerably more complex in construction. At the present time it is used to protect individual elements of the TPE against icing (for example, nose case ribs). The system is not widely used because of the limited thermal energy capacity;

heating of the engine compressor rotor blades by eddy currents (Foucault currents). This system can be used in the case of steel blades and certain compressor stage designs.

On some flight vehicles the propellers are protected against icing by a liquid antiicing system based on the principle of wetting the surfaces which are subject to icing with special liquids. Some of these liquids prevent adhesion of water droplets and ice crystals to the protected surface, others dissolve the ice crystals, forming a composition with freezing point lower than the ambient air temperature. Because of serious drawbacks (high liquid consumption, low effectiveness because of incomplete wetting of the protected surfaces, fire hazard), this protection technique has not been widely used.

Special silicone compounds which have very low adhesion with ice have been developed to protect the rotating powerplant parts (propellers, for example) against icing. Because of the low adhesion, the ice crust is thrown off the protected surface when it reaches a definite thickness. It has been suggested that these coatings in combination with heating can make it possible to achieve a reduction of the power required by the thermal systems on the flight vehicles fixed surfaces as well (the wing, tail, and so on).

In selecting the type of anti-icing system, we must consider its weight, the possibility of mounting the heating packet on the protected surface, the advantages of the particular packet in relation to other types of packets, the possibility of more economical utilization of the energy taken from the engine, the location of the energy sources relative to the surfaces being heated, and the degree of influence of the energy extraction on flight vehicle performance.

Prevention of Component Icing

During operation of fuel systems cases are observed of icing of certain components and parts located in the tanks and in the fuel feed lines. These include primarily the fuel filters, check valves, devices for ensuring fuel supply to the engines under negative load factor conditions, and the protective screens. The primary cause of icing of these components is crystallization of supercooled emulsified water droplets as a result of their encounter with the cold surface of the filter or other fuel system components.

As is well known, all grades of aviation fuels are hygroscopic, i.e., they are capable of absorbing moisture. The dissolved water content in the fuel depends on its chemical composition, temperature, and also the humidity of the surrounding air. The lower the molecular weight of the fuel and the larger the aromatic hydrocarbon content in the fuel, the less hygroscopic is the fuel. The dissolved water content in the fuel increases with increase of the air humidity.

When the ambient temperature decreases there is a reduction of the fuel temperature in the tanks. It has been found that during nonstop flights at an altitude of 7000-9000 m for a period of four or five hours, regardless of the time of year and ground air temperature, the fuel temperature in flexible tanks decreases to minus 15-20°C and in metallic tanks (for example, the integral tanks of the Il-18 airplane) to minus 35-40°C. As the fuel cools, the water content in the fuel cannot exceed its solubility at the given temperature, and the excess water separates from the fuel in the free state. As a result of the very small diameters of the water droplets (10-14 microns) and the high density of the cold fuel, the separating droplets are initially in the suspended state, distributed uniformly through the entire fuel volume. Under the action of the flight vehicle structure and fuel tank vibrations, the fine droplets coalesce into larger droplets, which settle and accumulate in the lower layer of the fuel, forming an emulsion (water-fuel mixture). The longer the flight duration, the more the fuel cools and, consequently, the larger the portion of the dissolved water which separates and appears in the fuel in emulsion form.

The presence of emulsified water in the fuel presents a serious hazard, since such water does not freeze for a long time at low temperatures and its droplets are in the supercooled state in the fuel. As this fuel passes through the engine feed line, the emulsified water droplets encounter the surface of the various components and are instantaneously transformed into ice, causing icing of the components and interfering with normal operation of the fuel system.

Thus, the icing process in fuel systems takes place similarly to airframe icing in flight, when the supercooled water droplets in the clouds crystallize instantaneously as they collide with the leading edge of the wing (empennage). We note that the formation of supercooled emulsified water droplets in the fuel takes place relatively rarely and only under favorable conditions. In most cases the water separating from the fuel freezes immediately, transforming into ice crystals.

Ice crystals may appear in the fuel as a result of frost falling from the tank walls. The amount of frost and therefore the amount of moisture which gets into the fuel will be greater the less fuel there is in the tanks, i.e., the larger the free internal tank surface. Condensation of moisture from the air as a result of temperature takes place not only on the tank walls, but also on the surface of the cooled fuel. Moreover, the moisture which condenses on the surface of the fuel does not freeze immediately; rather it initially spreads in the fuel and only after some period of time, when it is cooled markedly, does the process of its crystallization with the formation of ice begin.

Various techniques are used to prevent formation in the fuel of supercooled water droplets and ice crystals. The simplest technique is addition to the fuel of special additives which increase the solubility of the water in the fuel or which form with the water a mixture with low freezing point. In most cases substances which dissolve well in the fuels (alcohols, for example, tetrahydrofurfuryl, esters, and so on) are used as these additives. Among the special additives wide use has been made of ethylcellulose. When uniformly distributed through the entire fuel mass, ethylcellulose mixes with the water and forms an antifreeze with freezing point at minus 55-60°C. Depending on the ambient air temperature and flight duration, the addition of from 0.1 to 0.3% of this additive to the fuel is sufficient to prevent ice crystal formation.

A technique is available for periodic removal of ice from the fuel filters by brief application to the filters of small amounts of an antiicing liquid (ethyl and methyl alcohols, acetone). Control of the liquid feed is accomplished automatically by special sensors which activate when a given pressure drop across the filter is reached as a result of ice crystals clogging the filter. The time required to remove the ice from the surface of the filter when the system is activated does not exceed 20-30 sec. The drawback of this technique is that it does not eliminate the formation of ice crystals in the fuel and their deposition in other parts of the fuel system ahead of the filters.

Other methods used to reduce the amount of water dissolved in the fuel and prevent filter icing include: freezing-out, heating the filters in flight or briefly supplying warm fuel to the system, blowing dried air above the fuel surface, and centrifuging the fuel to separate the ice crystals.

Freezing water out of the fuel is accomplished in the wintertime at the fuel depots. To accomplish this the fuel is poured out into small surface containers and kept there for two or three days. This technique is very simple but requires the expenditure of time in additional filtering of the fuel. Heating of the fuel with hot air taken from the compressor or hot oil is widely used in foreign transport aircraft. The other protection techniques listed above require considerable energy expenditure and this has prevented their application to date under operational conditions.

Icing Indicators

In flight the crew can detect the initiation of icing by ice deposition on the individual parts of the flight vehicle, engine vibration, reduction of the flight speed, and so on. On some types of aircraft a visual icing indicator (Figure 137) is installed in the pilot's field of view for determining icing intensity. This indicator consists of a small profiled strut equipped with a probe, which is divided lengthwise into segments 10 mm long. This indicator can be used to evaluate the thickness of the ice layer and the icing intensity.

Figure 137. Probe type anti-icing indicator.

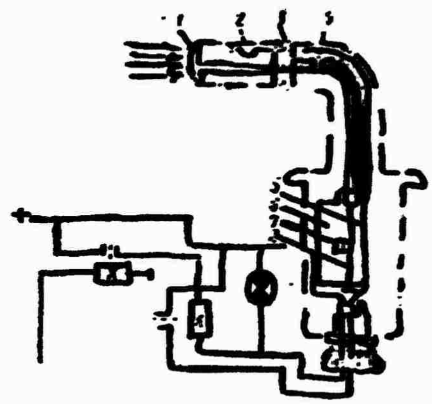

Figure 138. Membrane type icing warning system.

A membrane type warning device (Figure 138) can be used to warn the pilot of icing initiation. This device consists of two basic parts: a differential manometer with two sealed chambers 5, 6 and a combination total and static pressure pickup. The pressure pickup can be installed in the engine air intakes, wing leading edges, and other parts of the aircraft which are subject to icing. The operation of the warning system is based on utilization of the elastic properties of the sensitive element — the corrugated metal membrane 8, which closes and opens the electrical contacts 7 as the dynamic pressure of the airstream flowing over the warning system probe varies.

The total pressure of the relative airstream is fed through ports 1 to chamber 6. The chamber 5 receives static pressure through port 3, located on the side surface of the probe. In flight when there is no ice on the warning probe, the total pressure of the relative airstream is transmitted through ports 1 to chamber 6. Under the action of the pressure difference the membrane deflects and holds the contacts 7 in the open position. During flight in an icing zone the ports 1 are blocked by the ice film, and the total pressure is no longer supplied to chamber 6. The pressure in chambers 5 and 6 equalizes through the orifice 4, and the membrane returns to its undeflected position, closing the contacts 7 of the circuit which supplies current to the "Icing" warning light 9 and the warning system probe heater 2. Under the influence of the heat released by the heater, the ice melts and the ports in the probe become open. The restored pressure difference in the chambers breaks the contacts, thus turning off the warning light and warning system probe heater. If the aircraft has not left the icing zone by this time, the total pressure sensor is again subjected to icing and the cycle repeats.

Ice warning systems of the membrane type are used to monitor the initiation of helicopter main rotor blade icing. In this case the warning system includes pressure pickups, one located on each blade, the instrument case (differential manometer), and the electrical wiring and warning light. The instrument case has two cavities, separated by a membrane. One cavity is connected with the pressure pickups; the other is connected to the atmosphere. In addition to the dynamic pressure, which causes pressure increase in the instrument case in translational flight, the air in the lines from the pressure pickups to the instrument case is subjected to centrifugal forces, which tend to reduce the pressure in the instrument case. In the absence of icing these two forces balance and the pressure in the instrument case is approximately equal to atmospheric pressure.

During icing conditions the pressure pickup ports are covered with a crust of ice, the dynamic pressure input ceases, and under the influence of the centrifugal forces of the air the pressure decreases in the instrument case, connected with the atmosphere by lines through calibrated orifices on the blade trailing edges. Under the influence of this force the membrane deflects, activating the "Icing" warning light. As soon as the ice is thrown off the blades the system returns to the initial condition. The pressure reduction in the instrument case can be created by special injectors supplied with air from the engine compressor. Drawbacks of the membrane-type warning system are the delay in generation of the icing initiation signal and also the impossibility of measuring the icing growth rate and indicating the type of icing.

Among the other types of icing warning systems, use has been made of the radioactive warning system (Figure 139). The warning

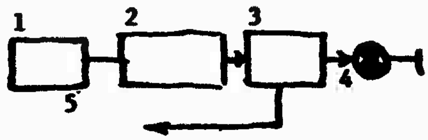

Figure 139. Block diagram of radioactive ice warning system: 1 - sensor; 2 - electronic unit; 3 - delay unit; 4 - warning light; 5 - to sensor heating element.

sensor includes a beta-radiation source (strontium 90 and yttrium 90) and a radiation counter. The electronic block provides 450 V power to the counter and amplifies the signal obtained from the counter. The delay block transforms the periodic signal into a continuous signal when the aircraft enters an icing zone.

The operating principle of the instrument is based on deceleration of the beta radiation by the ice layer which grows on an exposed cylindrical probe. The sensor operates as follows. The beta-particle flux, emitted continuously by the radioactive source, strikes the counter. The voltage pulses from the counter are recorded by the electronic unit. In the electronic unit the beta ray flux variation is transformed into a voltage variation. If the voltage exceeds a given level, a relay circuit is triggered and its contacts activate a warning light to alert the pilot that ice is forming. At the same time the sensor heating element is activated and this removes the ice from the probe surface. After the ice is removed, the instrument returns to the original condition. The process is repeated as long as the aircraft is in the icing zone. The ice thickness on the sensor probe at which the warning system operates is about 1 mm.

On some aircraft types, ice warning systems are installed which are based on the electrical conductivity of the ice which deposits on the sensor surface. When ice deposits on the sensor surface, the sensor contact rings close. Since the ship's voltage is applied to one of the rings, while the control grid of a thyratron is connected to the other ring, when the circuit is closed a voltage sufficient to trigger the thyratron is applied to its grid. This leads to triggering of the system, i.e., activation of heating for the sensor and lighting of the "Icing" warning light. The warning system servo unit includes circuits which prevent false triggering of the system during flight in rain and prevent premature disconnects and random circuit closure.

Worthy of examination is the warning system based on the following principle. The sensor is a cylindrical probe which is maintained in a state of axial vibration at the resonant frequency by a special generator. During flight in an icing zone, ice collects in the sensor passage and reduces its resonant frequency. A frequency change below some definite magnitude is sensed by the control system which transmits the "Icing" warning signal and activates the sensor heating system for several seconds. The heating time is chosen to be sufficient to remove the ice and prepare the sensor for the next operating cycle.

An icing sensor consisting of a rod which can be extended into the relative airstream and rotated by an electric motor has been used widely on foreign airplanes. The ice which forms during the icing process is removed by a metal knife. When the torque increases to a definite magnitude, the system triggers and activates the warning signal. By connecting the anti-icing system controls to the warning system, we can provide automatic control of the anti-icing system, relieving the crew of the necessity for performing excessive operations during flight and increasing the effectiveness of the aircraft icing protection.

Analysis

To ensure effective operation of the anti-icing system it must be supplied with an amount of heat which under the most severe weather conditions (with maximal possible value of the water content in the air for the given temperature) is sufficient to maintain the supercooled water which strikes the heated surface in the liquid state, transform the ice crystals or snow into water, and permit the incident airstream to carry the water away.

The thermal flux required for the continuous-duty system can be determined from the heat balance equation for the surface which is subject to icing. Neglecting secondary terms (the heat flux removed from the surface by radiation and thermal conduction), we can write the heat balance equation for the steady-state process in the form

$$Q_{AIS} = Q_{conv} + Q_{hw} + Q_{vap} - Q_{aero} \ \mathrm{W},$$

where Q_{AIS} is the heat flux supplied to the surface from the anti-icing system;

Q_{conv} is the heat flux removed from the surface by convective heat transfer;

Q_{hw} is the heat flux removed from the surface to heat the water droplets deposited on the surface to the surface temperature;

Q_{vap} is the heat flux removed from the surface as a result of vaporization of the water droplets deposited on the surface;

Q_{aero} is the heat flux supplied to the surface as a result of aerodynamic heating.

Replacing Q for each term of the equation by the specific thermal flux q, i.e., the amount of heat passing through a unit surface area in unit time, we obtain

$$q_{AIS} = q_{conv} + q_{hw} + q_{vap} - q_{aero} \ \mathrm{W/m^2}.$$

Let us define each term of this equation. In accordance with the Newton formula

$$q_{conv} = \alpha(t_{sur} - t_0),$$

where α is the heat transfer coefficient, $\mathrm{W/m^2 \cdot deg}$;

t_{sur} is the surface temperature, deg;

t_0 is the undisturbed stream temperature, deg.

Since the engine parts which are protected against icing are made in the form of smooth aerodynamic profiles, the average heat transfer coefficient for them can be calculated [54] using the formula

$$\bar{\alpha} = 5{,}5 \cdot 10^{-4} \frac{(\gamma_0 V_0)^{0.8}}{l^{0.2}},$$

where p_0 is the air pressure at the given altitude, N/m^2;

V_0 is the freestream velocity, m/sec;

l is the distance along the profile contour from the leading edge to the surface point in question, m.

For a cone in the limits of the distance l from the apex

$$\bar{\alpha} = 5{,}33\cdot 10^{-4}\frac{(\rho_0 V_0)^{0.8}}{l^{0.2}}.$$

The calculation of the local heat transfer coefficients for propeller blades is made using the following formulas.

For the leading edge and on the remaining surface of the profile, respectively,

$$\alpha = 9{,}35\cdot 10^{-4}\frac{(\rho_0 V_{r_0})^{0.8}}{D^{0.2}}; \quad \alpha = 4{,}4\cdot 10^{-4}\frac{(\rho_0 V_{r_0})^{0.8}}{l^{0.2}}.$$

where D is the diameter of a cylinder equivalent to the profile nose, m.

For air propellers

$$V_{r_0} = \sqrt{V_0^2 + V_r^2},$$

where $V_r = \omega r = \frac{\pi n}{30} r$ is the tangential velocity;

r is the distance from the axis of rotation to the blade segment in question, m;

n is the rotational speed, rpm.

The formulas presented above can be used to calculate α for any aerodynamic profiles with turbulent flow in the boundary layer (Table 6).

The specific thermal flux required to heat the water which strikes the protected surface is

$$q_{hw} = m_w c_w (t_{sur} - t_0),$$

TABLE 6. VALUES OF HEAT TRANSFER COEFFICIENT AS FUNCTION OF FLIGHT SPEED AND ALTITUDE, $W/m^2\cdot deg$

Flight altitude, m	Flight speed, km/hr		
	[illegible]	300	700
0	214	214	268
2000	200	230	256
4000	192	224	244
6000	172	200	227
8000	150	160	206

where m_w is the water mass deposited per unit surface per unit time, $kg/m^2\cdot sec$;

c_w is the specific heat of water, $J/kg\cdot deg$.

With account for the droplet deposition ratio (0.8), $m_w = 0.8\cdot10^{-3}wV$ (w is the water content of the air in grams per cubic meter).

In designing the anti-icing system we use the water content conditions shown in Table 7 for the corresponding ambient air temperature. Under short-time icing conditions (extent of icing zone 5-10 km), the systems must provide protection against icing for water contents 1.7 times those shown in Table 7. In analyzing engine anti-icing systems the airstream velocity may be determined from the fuel consumption using the relation

$$V = 4{,}15\cdot10^{-3}\frac{G_{в.пр}}{F},$$

where $G_{a\cdot red}$ is the reduced air flowrate, kgf/sec;
F is the inlet duct cross-section area, m^2.

TABLE 7. DESIGN ICING CONDITIONS

Ambient air temperature, °C	0	-10	-20	-30
Water content, g/m^3	0.8	0.6	0.3	0.2
Maximal flight altitude, m	5000	8000	9000	9000

For the known $q(\lambda)$ we use gasdynamic function tables to find the values of the velocity coefficient λ and then the airstream velocity in the inlet duct

$$V_{в} = \lambda a_{кр} = 18{,}3\lambda\sqrt{T_H^*}$$

The specific thermal flux required to evaporate the water striking the heated surface is

$$q_{исп} = 0{,}623\,\frac{L\alpha}{c_p}\cdot\frac{e_{t_{пов}} - e_{t_0}}{p_0},$$

where L is the latent heat of vaporization, J/kg;

- $e_{t_{sur}}$ is the water vapor pressure at the temperature t_{sur}, N/m^2;
- e_{t_0} is the water vapor pressure at the temperature t_0, N/m^2 (Appendix 9);
- c_p is the specific heat of air at constant pressure, J/kg·deg;
- p_0 is the ambient air pressure, N/m^2.

The specific thermal flux supplied to the surface as a result of aerodynamic heating is

$$q_{аэр} = \frac{k-1}{2kgR_{в}}\,\alpha r V_0^2 = \frac{\alpha r V_0^2}{2000},$$

where r is the temperature recovery factor.

For approximate calculations at low flight speeds, when aerodynamic heating of the surface is slight (the most severe conditions for operation of the anti-icing systems), without account for the heat required for heating the water

$$q_{AIS} = \alpha(t_{sur} - t_0)X,$$

where $X = 1 + \frac{1550\,(e_{t_{пов}} - e_{t_0})}{p_0\,(t_{пов} - t_0)}$ is the Hardy coefficient, which accounts for the increase of the overall thermal flux density relative to the convective heat transfer flux density.

The analysis of the continuous-duty thermal antiicing system (icing prevention system) is made for a given temperature differential, which is defined by the difference between the temperature of the surface being heated and the surrounding air for flight outside clouds. Studies [56] have shown that the required differential for aircraft profiles in use at the present time amounts to at least 50°C at sea level.

From the known values of q_{AIS} we can find the required air flowrate through the thermal chamber or the power of the electric energy sources. For the hot-air system the air flowrate is

$$G_{\text{в}}=\frac{q_{\text{пос}}S}{c_p(t_{\text{вх}}-t_{\text{вых}})},$$

where S is the surface area protected against icing, m^2;

t_{in} and t_{out} are respectively the air temperatures at the inlet and outlet from the thermal chamber, deg.

Since the heat utilization coefficient in the anti-icing system

$$\eta_{\text{т}}=\frac{t_{\text{вх}}-t_{\text{вых}}}{t_{\text{вх}}-t_{\text{пов}}},$$

the expression for the air flowrate can be written as

$$G_{\text{в}}=\frac{q_{\text{пос}}S}{c_p\eta_{\text{т}}(t_{\text{вх}}-t_{\text{пов}})}=\frac{\alpha S(t_{\text{пов}}-t_0)X}{c_p\eta_{\text{т}}(t_{\text{вх}}-t_{\text{пов}})}.$$

Specifying the temperature of the hot air at the inlet to the thermal chamber and the design surface temperature (it is usually assumed that $t_{sur} = 0$), we can calculate approximately the required air flowrate for the given icing temperature t_0 [54].

With increase of the flight speed to $V = V_{cr}$ the operating conditions of the thermal anti-icing systems deteriorate, and the thermal flux required to combat icing increases (Figure 140). This is a result of the fact that in this speed range increase of the flight speed leads to increases of the heat transfer coefficient, while the surface temperature increase owing to aerodynamic heating is small. With further increase of the flight speed, aerodynamic heating of the surface leads to improvement of the antiicing system operating conditions. For flight speeds of 600-700 km/hr and ambient air temperatures of minus 15°C or below, aerodynamic heating does not ensure protection of the aircraft against icing; however it may be used as an additional source of heat.

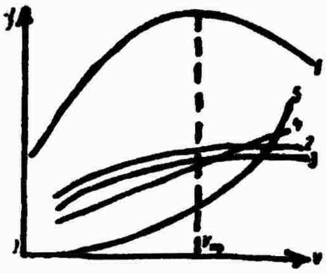

Figure 140. Specific thermal fluxes versus flight speed: 1 - q_{AIS}; 2 - q_{vap}; 3 - q_{conv}; 4 - q_{heat}; 5 - q_{aero}.

At supersonic flight speeds the danger of powerplant icing is practically eliminated as a result of aerodynamic heating. However, this does not mean that anti-icing protection is not required for supersonic airplanes. The takeoff, climb, descent, and landing of these airplanes are associated with speeds at which the probability of icing is just as great as for the subsonic airplanes. Therefore, no matter how short the time the supersonic airplane flies at subsonic low speeds, it does require an anti-icing system.

Operation

Maintenance of the anti-icing system does not cause any particular difficulties. System operation is checked at the intervals prescribed by the maintenance manual, after long-time airplane storage (more than two months), and after performing operations associated with disassembly of the individual antiicing system components. If the aircraft is equipped with a hot-air system, it is checked on the ground with the engines operating from 20% rated power up to rated power. The system operation is monitored by the lights which indicate opening of the air inlet valves, or by the temperature of the air entering the system. Check of the air intake antiicing system can be made with the engines not running by visual check of the travel of each control valve from the closed to the open position and back. Here we must make certain that the valve and its electrical drive operate smoothly, without sticking, and there is no play in the valve control rods.

The check of the electro-thermal anti-icing systems can be made either from the ship's generators (with engines running) or from the ground power units. The system operation is monitored by the ammeters and the time the warning lights come on and go off. To avoid overheating, which could result in possible deformation of the heated surfaces, the anti-icing systems are activated on the ground for a time not exceeding that listed in the operating instructions for the given aircraft type. If in this process the warning light operation is not correct or the ammeter indications do not correspond to the established values, the system must be turned off and the programmer contacts and the heating elements must be checked. The condition of the anti-icer external surfaces is verified by visual inspection, which involves checking for the absence of mechanical damage, skin wrinkling, burned areas, and other defects.

Below-freezing ambient air temperatures and high water content lead to ice formation in the engine air intake while the aircraft is parked. Moreover, because of the small radial clearance between the blades and the compressor (turbine) casing, cases have occurred in which the blades froze to the case. Therefore, prior to starting the engine, while preparing the powerplant for flight, an inspection must be made of the engine inlet duct, the guide vanes, and the first compressor stages. If ice, snow, or frost is found on these surfaces it is recommended that the inlet duct be heated and dried using hot air.

Engine starting under icing conditions with the electro-thermal anti-icing system turned on does not present any danger, since the heated surfaces reach the design temperature quite rapidly. Because of slow warmup of the protected surfaces, the hot-air anti-icing system does not eliminate the possibility of ice formation on these surfaces during the engine starting process. Ice formation on the inlet components will terminate when idle engine speed is reached.

For some engine types with low compressor pressure ratio, the thermal energy at idle or near idle conditions may not be sufficient to eliminate ice formation. In this case it is recommended that the engine operating regimes which are hazardous with regard to icing be limited or avoided completely.

Flights in icing conditions must be conducted with the engine and propeller anti-icing systems on at all times. The systems should be turned on prior to entering the icing zone and turned off after the aircraft leaves this zone. We note that the electro-thermal system has considerably less thermal lag than the hot-air system, and this means that the hot-air system must be turned on early when the possibility of icing is present, so that the protected surfaces can warm up properly.

CHAPTER 7

FIRE PROTECTION SYSTEMS

Fire Hazards

The possibility of fire occurring in the process of aircraft operation is the result of the following factors:

presence on board the aircraft of large quantities of combustible materials;

self-ignition of the fuels and oils if they come into contact with the hot surface of the engine or powerplant components;

location of the fuel tanks close to the operating engines;

various sorts of flight incidents and emergency situations resulting from failure of individual components, violation of fire safety regulations and instructions on aircraft operation and maintenance;

explosion of fuel vapors in the tank free space or in any other closed volume when a flame occurs in such a zone.

Fuel ignition sources may be the result of damage to individual segments of an electrical conductor or discharges of static electricity formed as a result of friction between the aircraft skin and air during flight, and can also occur during refueling as a result of fuel friction as it flows in the refueler hoses.

The fire hazard of the various fuels depends on their chemical composition and the existing conditions. It has been established that under the same conditions the heavier fuels ignite more easily, since they oxidize more easily and form explosive mixtures. Thus, kerosenes ignite at a temperature 100°C lower than the gasolines. The hydraulic fluids and lubricating oils ignite even easier. In the case of an aircraft accident they form the first fire sources, and then the fuel is ignited from them.

Fuel will ignite from an outside source only when a mixture of fuel vapors and air which is capable of burning is formed above the fuel. For aviation fuels at atmospheric pressure the lower ignition limit (the leanest mixture capable of burning) corresponds to an excess air coefficient of 1.4 while the upper limit (the richest mixture capable of burning) corresponds to an excess air coefficient of 0.5.

The formation of explosive mixtures depends on the fuel temperature. There are lower and upper temperature limits for the formation of explosive mixtures. As the lower limit we take that minimal fuel temperature for which its vapor pressure reaches magnitudes at which an explosive mixture is formed in the closed tank space. Upon further cooling of the fuel the mixture becomes so lean that ignition is difficult. The maximal fuel temperature at which the mixture retains its explosive properties is taken as the upper limit.

Reduction of the possibility of fire occurring in the aircraft, rapid fire localization, and extinguishment are accomplished by use of design techniques, installing fire protection equipment, filling the tanks with inert gas, and improving the effectiveness of the fire fighting equipment and reliability of the warning system.

Design Measures Directed to Ensuring Fire Safety

One of the primary measures ensuring flight safety is the creation in the aircraft of conditions under which the possibility of fire occurring and spreading is prevented. Measures taken in nacelle design with these objectives include installation of sealing partitions, use of high-temperature materials for the lines running in the engine hot-section zone, locating the flexible connections outside the firewalls, and use of high temperature insulation for the wire bundles.

The components of the fuel, oil, and hydraulic systems are usually located in the powerplant cold zone and as compactly as possible. The use of electron (a magnesium base alloy) and other combustible materials which are difficult to extinguish is not permitted in the hot zone. All the highly heated engine parts and components are cooled. The air intakes and the outlet slots in adjacent powerplant compartments are staggered relative to one another in case of fire. Provision is made for rapid engine shutdown and propeller feathering on PE and TPE when an engine emergency situation is indicated.

A fire shutoff valve which blocks fuel flow to the engine is installed in the engine fuel feed line. The valve location is selected to minimize the amount of fuel which can be used by the engine after closing the valve. The fuel lines are located in the powerplant firesafe zone and are protected as much as possible against damage in case of an emergency landing.

It is advisable that the engines be located on pylons below the wing or on the aft fuselage. The engine attachment to the pylon can be accomplished so that they can be jettisoned if necessary. To reduce the probability of fire during an emergency landing it is desirable that the airplane have a high wing or engines located above the wing. The fuel tank groups located in the wing must be separated from one another by partitions, and the spars, ribs, frames, and containers in these regions must have sealed webs. The oil tanks are located in the nacelles so that in case of leakage the oil will not come into contact with hot engine parts.

With regard to configuration of the fuel and oil systems, any possibility of accumulation of combustible liquids or their vapors must be eliminated. The outlets of the lines for dumping fuel in flight are located out of the hot exhaust gas zone. Provision is made to block the air flow through the nacelle by complete closure of the flaps which control the air flow through the region inside the cowling and closure of the engine air bleed straps (valves) when the fire extinguisher system is activated. Electrical equipment located in regions of possible fuel vapor accumulation must be of explosion proof construction. Provision must be made in the electrical system for switches which make it possible to cut off in case of fire the electrical supply sources and the segments of the electrical system which are not critical to continuation of the flight and landing. To reduce the possibility of fire occurring in the cabin, most of the furnishings are fabricated from noninflammable materials.

In aircraft with the powerplants located in the immediate vicinity of the cockpit or the passenger cabin, partitions are installed to separate the powerplant from the cockpit and cabin and prevent spread of fire and smoke into these areas. At the same time the cabins are equipped with emergency exits to permit rapid egress from the aircraft.

Among the other design steps taken to ensure fire safety, we note the use of systems for monitoring the most important engine operating parameters and the systems for warning of engine operating problems, the installation of thermal relief valves in the fuel systems, provision for bonding of all parts of the aircraft, installation of reliable grounding and brush-type static electricity dischargers, and the installation of devices to block entry of air into the pressurized cabin in case of powerplant fire.

Fire Protection Equipment

The fire protection equipment is fundamentally the same on all modern aircraft. It includes the warning systems, fire extinguishing systems, and portable extinguishers. The warning systems are designed to provide early fire detection, warn the crew, and activate automatically the first bank of the fire extinguishing system. The fire extinguishing systems are provided to deliver the extinguishant from the bottles to the fire source.

Reliability of the warning system can be provided if the following requirements are satisfied:

rapid action; the time from appearance of the fire until the signal is transmitted must not exceed three sec;

repetitive action and reliability of operation under all operating conditions;

rapid return of the system to the original status after termination of the fire;

absence of false signals;

operability of the warning sensors during vibration and other loads which arise in operation; the sensors must not be sensitive to contact with oil, fuel, water, or hydraulic fluids;

operational simplicity, possibility of verifying operability of the individual system elements prior to flight without removing them from the aircraft.

Special detectors serve as the sensitive element of the warning system. On the basis of operating principle, they can be subdivided into two groups. The first group of detectors reacts to the maximal temperature, while the temperature at which the detectors of the second group trigger depends on the rate of temperature rise (differential detectors). The first group includes the semiconductor, bimetallic (membrane), and certain other detector types.

The semiconductor detectors consist of a metallic capillary tube inside which there is located a conductor (small rod). The space between the conductor and the tube wall is filled with a semiconductor material whose electrical conductivity increases as the temperature increases. The central conductor is one of the arms of a balanced bridge in the recording instrument. Occurrence of fire and heating of the detector to a definite temperature causes reduction of the semiconductor material resistance, as a result of which the bridge balance is disturbed and a current appears in the bridge diagonal. This leads to closure of the electrical circuit and activation of the system, i.e., turning on the warning light and actuating the fire extinguishing system.

In the bimetallic detectors the sensitive element is a membrane made from two materials which have different linear expansion coefficients. Most frequently use is made of invar (64% iron and 36% nickel) with a linear expansion coefficient of $1.5 \cdot 10^{-6}$ mm/deg·m and tombac (90% copper and 10% zinc), whose linear expansion coefficient is $18 \cdot 10^{-6}$ mm/deg·m. When a temperature of 140-170°C is reached in the zone where such detectors are located, the membrane deflects, thereby providing actuation of the fire warning and extinguishing systems. The application of these detectors has been limited recently because of the high inertia of the detectors, and also the possibility of false triggering in case of shorting of the detector circuit.

Detectors are being developed which consist of flexible metal tubes filled with an inert gas of high absorptivity. Inside the tube are a special core and an inert filler of high thermal conductivity. The core material has the property that at low temperatures it absorbs the inert gas, and upon increase of the temperature it actively releases the inert gas, thereby increasing the pressure inside the tube, which leads to closure of the detector circuit and activation of the warning system.

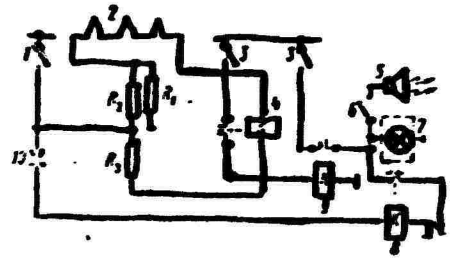

Figure 141. Fire warning system with differential-type detectors: 1,3,6 - switches; 2 - fire warning sensor; 4,8, 9 - relays (contactors); 5 - siren; 7 - warning light; 10 - diode; R_1, R_2, R_3 are resistors.

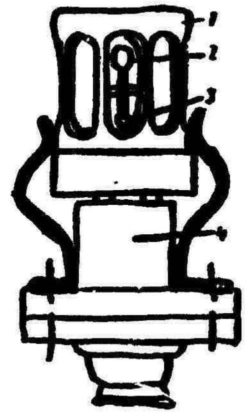

Figure 142. Detector: 1 - case; 2 - working (hot) junction; 3 - cold junction; 4 - socket.

The sensitive element of the system with differential detectors (Figure 141) is the thermopile 2, assembled from thermocouples connected in series (chromel-kopel, for example). The fast-response thermocouple junctions (disks of diameter 3 mm and thickness 0.16 mm) are located in the upper part of the sensors, while the slow-response junctions (balls formed by welding the other two ends of the electrodes together) are located in the lower part (Figure 142). If there is a rapid temperature rise the fast-response junctions heat up far faster than do the slow-response junctions, an emf appears at the output of the sensor, is amplified, and is used as the fire warning signal. The moment at which the warning system triggers depends on the absolute value of the temperature and the rate of temperature rise in the zone where the sensors are located, i.e., increase of the rate leads to triggering of the system at lower temperatures. For example, for the SSP-2 system, which is presently installed on Soviet aircraft, for a temperature rise rate of 2 deg/sec the system triggering temperature is 220°C, while for a rate of 10 deg/sec the triggering temperature is 165°C.

The second component part of the warning system is the actuating unit, which includes the low-resistance polarized relay 4 (see Figure 141), whose contacts close the relay 9 of the fire signaling and extinguishing circuit; the relay 8, used to check the system integrity; the resistor R_1, used to limit the current in the polarized relay when checking system integrity; and the resistors R_2 and R_3, used to calibrate the resistance of the sensor detector circuit when selecting a given triggering temperature.

If there is a change of the temperature of the medium surrounding the detectors at a given rate and if the detectors are at the same time heated to their triggering temperature, the detector emf develops a current in the winding of the polarized relay 4 which is sufficient to activate the relay. The contacts of relay 4 close and activate relay 9. The latter activates the visual and audio warning signals and causes automatic discharge of the first bank of extinguishers.

Fire extinguisher systems are subject to the following requirements:

reliability of operation in all flight regimes and altitudes, possibility of directing the extinguishant from each extinguisher to any protected compartment;

maintenance of the extinguishing agent concentration in the compartments for no less than 3-5 sec;

rapid activation and effectiveness of the extinguishing action; bottle discharge time must not exceed 5-7 sec;

extinguishant supply in system sufficient for two or three applications, with the first bank of extinguishers being activated automatically by the fire warning system, and the second and third banks being activated manually;

location of the bottles with the extinguishing agent and the lines connecting them with the spray manifolds in areas which are most protected against possible damage in an emergency landing;

presence of simple and reliable methods for checking the system operating condition; possibility of monitoring the pressure in the bottles during maintenance of the fire protection system;

rapid preparation for operation when servicing after discharge of the system; convenience of inspection, assembly, and disassembly of the individual components;

possibility of automatic activation of all extinguishers simultaneously to supply the agent to the compartments with the greatest fire hazard in case of forced landing with the gear up.

Let us examine the operation of the system for extinguishing a fire in the engine nacelles (Figure 143). When a fire develops in the zone where the detectors are located the latter send an electrical signal to the fire extinguishing system. This signal activates the solenoid valve unit 3 to supply the fire extinguishing agent to the engine on which fire has developed in the cowled region. A limit switch closes the circuit for lights indicating the open position of the distribution valve and fires the cartridges of the first bank of extinguishers. The agent flows through the open valve and the spray manifolds, fills the space inside the engine cowling, and thus prevents atmospheric oxygen from getting to the flame source. If the fire is not put out after activation of the first extinguisher bank, the extinguishers of the second and third banks are activated. If a fire is discovered visually the first extinguisher bank can be activated manually.

Special inertial or pressure switches which activate at definite load factors (inertia switches), or when they are depressed in case of landing with the gear up, are used to activate the fire extinguishing system in emergency cases.

The amount of agent G_{ext} of the first bank required to put out a fire can be calculated approximately from the formula

$$G_{ext} = G_{air}\,\alpha_{ext}\,\tau,$$

where G_{air} is the maximal air flowrate through the space inside the engine cowling, kgf/sec;

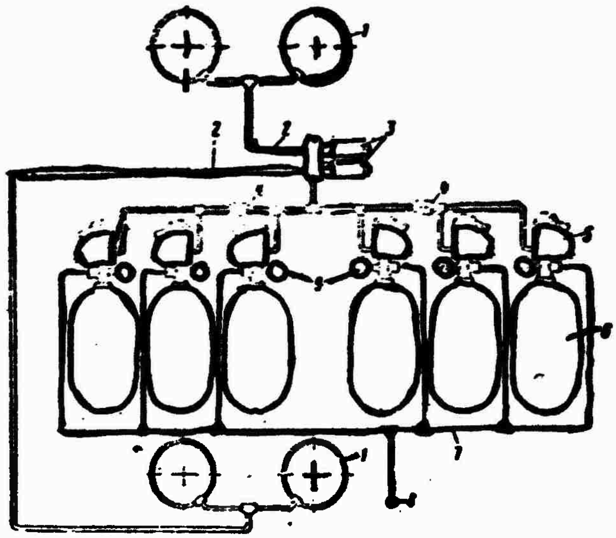

Figure 143. Fire extinguishing system in engine nacelles: 1 - manifold; 2,7 - lines; 3 - electromagnetic valve block; 4 - check valve; 5 - head/stop-valve; 6 - bottle; 8 - visual discharge indicator; 9 - pressure gages

α_{ext} is the required agent concentration;

τ is the time during which the required agent concentration is maintained in the cowled space, sec.

According to [47] the consumption of the "3.5" agent composition does not exceed 250-260 g/m^3. If there are significant air currents in the cowled engine space the extinguishing agent supply of each section is increased by a factor of two or three. After determining the weight of the required amount of agent and knowing the bottle volume and filling factor, it is not difficult to calculate the number of bottles in each bank and then in the system.

The diameter of the branch line by which the bottle is connected to the main line is equal to the effective diameter of the port in the stop-valve (for most head types this is 10 mm). Equating the main line area to the sum of all the branch lines to the bottles, we can write

$$d_{line} = d_{br}\sqrt{n}$$

where d_{br} is the branch line diameter;

n is the number of branch lines, equal to the number of bottles per bank.

To improve the effectiveness of the fire extinguishing system, the agent flow can be combined with closure of the bleed strap (valve) and also with termination of the cooling air flow. A complete evaluation of the system effectiveness and capability is obtained after conducting special full-scale fire tests.

Fire extinguishing systems are being developed based on filling the space inside the engine cowling with a thermally insulating porous mass which absorbs the oxygen from the air. This mass is formed when liquid components, one of which is a synthetic resin and the other a 20% solution of sulfuric acid, are sprayed through nozzles into the space inside the cowling. The ratio of these components is about 4:1.

On some types of aircraft fire extinguishing systems for the interior engine cavities (nose case, gearcase, bearing zones, and so on) are provided in addition to the systems which supply extinguishing agent into the space inside the cowling. The systems may be either centralized or individual for each powerplant.

The electrical part of these systems is analogous to that examined above. The fast-response thermopile junctions are located inside the engine and the slow-response junctions are located outside. If a flame comes into contact with the junctions and the specified temperature is reached, an emf develops in the thermopiles which is sufficient to actuate the polarized relay of the actuator unit. The latter transmits a signal to light the warning lamp (red) on the instrument panel or on the fire extinguisher control panel in the cockpit and to activate automatically the fire extinguishing system in the engines.

When the cartridge fires, the force of the explosive gases opens the stop-valve and at the same time automatically activates the first bank of extinguishers of the primary extinguishing system; the agent flows into the internal engine cavities and through the distribution manifolds to the nacelle compartment of the same engine. The selection of the internal engine cavities into which the agent is supplied is determined by the probability of fire occurrence in these cavities. The optimal conditions for fire extinguishing are considered to be those in which the agent enters simultaneously into the fire-hazardous oil cavities and the adjacent air cavities.

Automatic introduction of the agent into the internal cavities may be ineffective if the engine is not shut down when the fire extinguishing system is activated, i.e., if the causes of the fire and the conditions favoring its development are not eliminated.

The effectiveness of the fire extinguishing system is to a considerable degree determined by the effectiveness of the extinguishing agents used. The basic requirements imposed on these agents include: high extinguishing effectiveness, stability of properties during operation and extended storage, safety in handling, and absence of harmful effects on the materials used.

For many years aircraft fire extinguishing systems used as the extinguishing agent liquid carbon dioxide, from each kilogram of which under normal atmospheric conditions there are formed 506 liters of carbon dioxide gas. The latter liquifies at a temperature of 0°C under a pressure of 35.5 kgf/cm^2.

In the filled cylinders, part of the liquid carbon dioxide changes into the gaseous state. If the carbon dioxide is released from the cylinder while holding the latter with the valve up, the carbon dioxide will come out in the form of a gas. This technique is used in inert gas systems for supplying carbon dioxide gas into the fuel tank air space. However, if the carbon dioxide is released from the cylinders under the pressure of its own vapors after first installing a siphon tube or holding the cylinder with the valve down, the carbon dioxide will be discharged in the liquid state (in the form of snow flakes). These flakes, vaporizing on the hot surfaces, create a high gas density in the combustion zone, lowering the percentage oxygen content to the point where it can no longer support combustion; the flakes also reduce the temperature and if the amount of gas is sufficient will make further combustion impossible.

The advantage of carbon dioxide in comparison with the other extinguishing agents is that it is not harmful to any of the materials subject to its action. Its disadvantages are the high required extinguishing concentration (23.5% by volume) and its reduced effectiveness at low temperature. The latter is explained by the fact that the saturated vapor pressure of carbon dioxide decreases markedly as the temperature decreases, which leads to slow discharge of the carbon dioxide from the cylinder if the system is activated at temperatures below freezing. Thus, at a temperature of minus 40°C the discharge time of the OSU-4 extinguisher (charge weight 5.7 kgf) is 11 sec, while at a temperature of plus 40°C the time is 5 sec (Figure 144). If we consider the line resistance during cylinder discharge, the discharge time at temperatures below freezing reaches 15-18 sec.

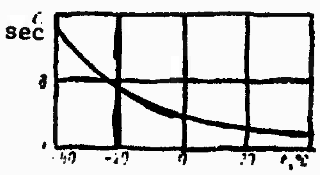

Figure 144. Extinguisher discharge time versus temperature.

At the present time the "3.5" composition, consisting of 70% (C_2H_5Br) and 30% CO_2, is widely used as an extinguishing agent. Pure ethyl bromide vapors when present in air to the extent of 6.75-11.25% (by volume) will ignite under the influence of a powerful pulse (electrical arc, white-hot metal spiral, and so on). A mixture of ethyl bromide and air will not ignite from the flame of gasoline, alcohol, and other combustible materials. Nevertheless, because of the noted problems ethyl bromide in the pure form is not used as an extinguishing agent.

The "3.5" composition will not ignite from any ignition source. Under standard conditions this is a liquid with specific weight 1.28 kgf/liter and freezing point below minus 60°C. This composition obtained its name from the fact that its required extinguishing concentration (6.7% by volume) is 3.5 times less than that of carbon dioxide. The "3.5" composition operates with nearly the same effectiveness over the temperature range ± 60°C.

The drawbacks of the "3.5" composition include the corrosive activity of ethyl bromide with respect to the aluminum and magnesium alloys and its toxicity. Chloroform ($CHCl_3$) in the amount of 4.2% of the ethyl bromide weight is added to the latter to protect these materials against corrosion. Since the saturated vapor pressure of the "3.5" composition is not large, the extinguishers must be charged with nitrogen to 85-90 kgf/cm^2 to increase the agent discharge pressure. When the "3.5" composition is delivered into the space inside the cowling, it vaporizes rapidly, forming a mixture of ethyl bromide and carbon dioxide gas vapors. Under standard atmospheric conditions about 150 liters of carbon dioxide gas and 140 liters of ethyl bromide are formed from each liter of the composition.

The "7" composition, containing 20% ethyl bromide and 80% methylene bromide (CH_2Br_2), is used to eliminate flame sources in the internal engine cavities. The specific weight of this liquid at a temperature of 20°C is 2.51 kgf/liter, the boiling point is 38-98°C, the freezing point is below minus 70°C, and the extinguishing concentration is 3% (by volume).

Among the other extinguishing agents, the most promising are dibromtetrafluorethane (CF_2Br-CF_2Br) or freon 114B_2. Its required extinguishing concentration is about three times less than that of the "3.5" composition. Under normal conditions this is a colorless liquid with specific weight 2.18 kgf/liter, boiling point 47°C, and freezing point minus 112°C. In addition to bromine, freon 114B_2 contains fluorine, which reduces to a considerable degree the corrosivity of the bromine. Freon does not react with the aluminum and magnesium alloys. If it gets into the engine oil cavities in case of inadvertent discharge of the extinguisher it has no harmful effect on the physical and chemical properties of the oil. Moreover, freon is operationally more convenient and simpler, since it is a single-component composition which is ready for use.

Inert Gas Systems

On modern aircraft, which carry large amounts of fuel, a considerable quantity of fuel-air mixture which is hazardous with respect to explosion and fire is formed in the tanks as the fuel is consumed. Short-duration (emergency) inert gas systems are installed to prevent explosion of this mixture on certain types of aircraft with unfavorable location of the fuel tanks (for example, in the lower part of the fuselage). Such systems provide flow of an inert gas into the tank air space prior to an emergency landing. The need for continuous-duty inert gas systems for use in flight on the supersonic airplanes arises because of the fact that with increase of the flight speed the temperature of the tank walls may reach the self-ignition temperature of the fuel vapors.

Nitrogen or carbon dioxide, for which the concentration required to provide explosion protection is 30-35% (in percent of the air volume), is used as the inert gas. The fire extinguishing effect of these gases lies in the fact that the oxygen concentration in the fuel-air mixture becomes less than the minimum at which combustion is possible. To ensure protection against explosion, the free oxygen concentration in the fuel tanks should not exceed 8-10% by volume.

In designing the inert gas system, the following requirements are imposed:

prevent ignition of the fuel-air mixture in the tanks under the various flight conditions. To satisfy this requirement, the specified volumetric concentration of the neutral gas must be maintained for the various fuel flowrates and flight regimes. The gas consumption must not be excessive when using an open venting system;

adequate supply of inert gas to create the required concentration in the tank air space throughout the entire flight. With the system operating, the pressure in the tanks must not exceed the acceptable value;

provision for instruments to monitor system operation in flight and check it out on the ground;

reliability of system operation throughout the entire temperature range in which the airplane is operated;

low inert gas solubility in the fuel;

low system weight, automatic actuation, operational simplicity and safety.

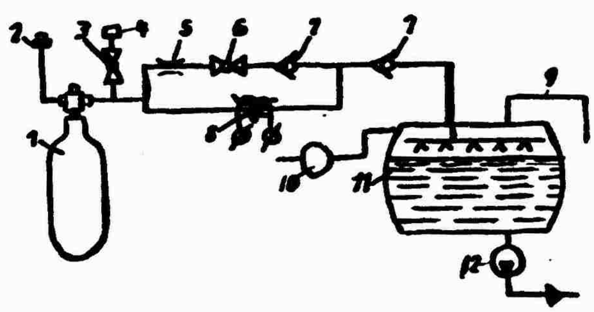

Figure 145. Inert gas system: 1 - bottle; 2 - visual discharge indicator; 3 - bleed valve; 4 - bleed fitting; 5 - orifice; 6 - shutoff valve; 7 - check valves; 8 - heated orifice; 9 - tank vent; 10 - pressure warning light; 11 - fuel tank; 12 - boost pump.

The inert gas system operates as follows (Figure 145). Prior to forced landing of the airplane with gear retracted or in case of fire in compartments located adjacent to the fuel tanks, the valve 6 is opened and gas flows from the bottle 1 into the tank air space. When the pressure in the tanks reaches 0.15-0.2 kgf/cm^2, the pressure sensor 10 activates and sends a signal to close the shutoff valve. At the same time the valve 3 is opened and the remaining inert gas is discharged into the atmosphere. The system also provides for the possibility of inert gas entry into the tanks through the heated orifice 8 throughout the entire flight. The gas pressure in the fuel tanks is about 0.12-0.15 kgf/cm^2. The calibrated orifice ensures the required pressure, specified gas flowrate, and bottle discharge time.

The inert gas flowrate W_{ig} depends on the differential pressure and is determined from the formula

$$W_{ig} = \frac{\pi d_{ж}^2}{4}\mu_{ж}\sqrt{\frac{2g\Delta p}{\gamma_{н.г.}}}\ \text{m}^3/\text{sec},$$

where d_{or} is the orifice diameter, m;

μ_{or} is the orifice discharge coefficient;

Δp is the differential pressure, kgf/m^2;

γ_{ig} is the specific weight of the inert gas, kgf/m^3.

If necessary, the inert gas can be used in the fire extinguishing system. For this purpose a combined stop-valve is installed on the bottles and is equipped with pyrotechnic cartridges for releasing the inert gas into the tanks or into the fire extinguishing system.

Approximate calculations of the inert gas supply are made using the formula

$$G_{ig} = \frac{\alpha_{ig}E_t\gamma_{ig}}{100}\ \text{kgf}$$

where α_{ig} is the required explosion-proof inert gas concentration, %;

E_t is the tank volume, m^3.

The drawbacks of this system include the considerable weight of the bottles and the necessity for maintaining a constant bottle temperature regime because of the marked variation of the pressure in the bottles when the temperature changes. Moreover, with increase of the fuel tank volume there is a considerable increase of the system weight, and regulation of the gas flowrate into the fuel tanks becomes complicated.

The systems in which the inert gas consists of fuel combustion products after removal of the water vapor may find application in

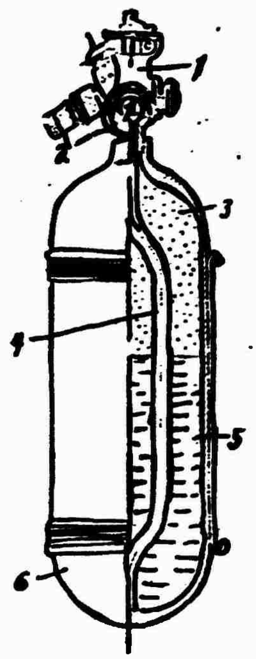

Figure 146. Bottle: 1 - head/stop-valve; 2 - pressure gage; 3 - compressed gas; 4 - siphon tube; 5 - extinguishant; 6 - body of bottle.

supersonic airplanes. Such systems usually consist of generators in which fuel mixed with air is burned to obtain the concentration corresponding to the inert gas composition, a condenser used for cooling the inert gas and partial removal of the moisture, a moisture freezing unit, and a dryer. These systems are not yet widely used because of poor regulation of the fuel and air supply to the combustion chamber during climbs in altitude, incomplete removal of moisture by the dryer, and unreliable starting.

Components

Extinguishers. The OS-8 extinguisher bottle (Figure 146) is a steel vessel with a cylindrical section having double brazed wire armoring. Inside the bottle there is a siphon tube through which the agent can be completely discharged. The upper portion of the bottle terminates in a neck into which the head/stop-valve is screwed (Figure 147). The head includes the valve, pyrotechnic, and an indicating-relief device.

The extinguisher is activated as follows. When the electrical circuit of the fire warning and extinguishing systems closes, the cartridge 2 explodes, the hot gases drive the piston 3, which lifts the lever 11. As the lever is raised, its shaft 4 rotates and allows the free end of the lever 5 to travel upward through the slotted part of the shaft 4 and open the stop-valve. Under the action of the gas pressure in the bottle and the spring compression force, the stop-valve poppet opens and allows the agent to escape from the bottle. The agent flows through fitting 10 to the solenoid valve and then to the

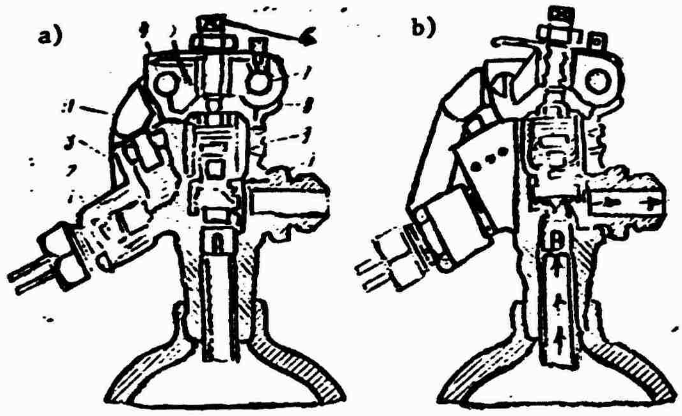

Figure 147. Head/stop valve: a - stop-valve closed; b - bottle discharge into system; 1 - squib; 2 - cartridge; 3 - release piston; 4 - latch-shaft; 5 - pivoting lever; 6 - adjustment screw; 7 - shaft; 8 - cap travel limiter; 9 - poppet; 10 - fitting; 11 - lever.

fire location.

The indicating-relief device is necessary to prevent bottle failure caused by increase of the pressure in the bottle above the limiting value. The operation of the device is as follows: if the pressure in the bottle increases to 200 ± 20 kgf/cm^2 the safety membrane ruptures and the extinguishing agent is discharged overboard through a line. The exit of this line is usually covered by a red plug (visual indicator), whose rupture indicates that the bottle has been discharged. The extinguishers are located in the central part of the fuselage or in the engine nacelles.

Portable hand CO_2 extinguishers of the OU type are used to extinguish fires which may occur inside the aircraft or in the power-plants during servicing. The weight of the charged extinguisher is 6.2 kgf; the CO_2 charge weighs 1.7 ± 0.1 kgf. The total volume of the extinguisher is 2.3 liters. On the bottle neck there is installed a stop-valve with a poppet which is connected by a discharge tube with the nozzle. The trigger which controls the extinguisher is located on a lever mounted on nozzle.

From three to six bottles are usually located on board the aircraft. They are mounted in the vertical position with the aid of two brackets. The base of the bottle rests on the lower bracket while the bottle is held against the upper bracket by a spring latch. To activate the extinguisher it is removed from the bracket, the nozzle is pointed toward the flame source, and the trigger is fully depressed. It is recommended that the discharged carbon dioxide be initially directed at the edge of the fire, gradually covering the entire area of the burning material.

Electromagnetic valves and check valves. The valve unit is designed to direct the extinguishing agent to the compartment where the fire is located. This unit consists of a housing into which there are threaded one inlet and several (one for each valve in the unit) outlet fittings. Each valve has a poppet and a solenoid unit (Figure 148). The solenoid unit is located in the housing 1 and consists of an electromagnet and core. Above the electromagnet there is a relay consisting of two switches. The switches and the plunger which connects the core with the relay are located outside the solenoid housing and are covered by the protective hood 4, which is anodized red.

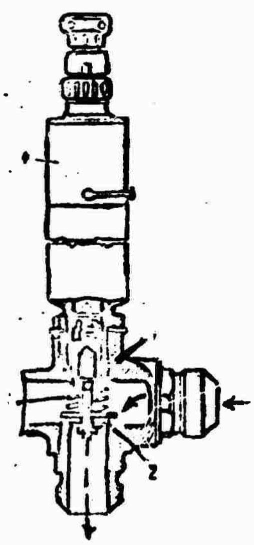

Figure 148. Electromagnetic valve.

When the electromagnet is energized the core displaces, overcomes the resistance of the spring 3, and lifts the poppet 2, permitting the extinguishing agent to flow to the fire location. At the same time, as the core displaces its free end strikes a rod connected with the relay switch plunger. This then closes the circuit linking the relay with the cartridge, the valve interlock system, and the fire warning system. When the electromagnet is de-energized the poppet closes and the core returns to its original position under the influence of the spring 3. The plunger also returns to its original position under the spring action and the relay contacts open.

Check valves prevent flow of the extinguishing agent into the discharged bottles when the next bank of extinguishers is activated. The check valves are of the disk type and their sealing is provided by lapping the disk to the valve seat. The disk is held against the seat in the closed position by its own weight.

Spray manifolds and lines. The spray manifolds are designed to scatter the agent in the fire zone. The agent is fed to the manifolds along lines from the electromagnetic valve units. The manifolds consist of perforated tubes with orifice diameters ranging from 0.6 to 1.2 mm. The number of orifices is selected on the basis that the total orifice area equal or exceed the manifold diameter by a factor of 1.2-1.5. To improve the spray pattern the orifices are staggered in the manifold tubes, i.e., they are directed along the engine cooling air flow, perpendicular to the cooling flow, and opposite the cooling flow. Statistical data indicate that the ratio of the manifold tube length to diameter is about 250-300.

The lines and components of the fire protection system are made from steel or aluminum alloys and are colored red. Flexible hoses may be installed in certain parts of the system. Steel lines should be used in systems with the "3.5" composition, and also in those areas where high pressure or temperatures develop in the lines. These areas include the lines connecting the bottles with the electromagnetic valve units and certain areas in the engine compartments. Standard fittings which provide a rigid leakproof joint are used to interconnect the lines. The lines and components of the fire extinguishing system are checked for leakage after assembly under an air pressure of at least 50 kgf/cm^2.

Operation

Operational reliability of the fire protection systems is ensured by constant surveillance of the systems during preflight, postflight, and periodic inspections. Particularly important is the check of the amount of agent in the bottle. If the charge weight is low, there will not be sufficient agent to put out a fire. Increase of the charge weight above the permissible value may lead to spontaneous discharge of the bottles if the ambient temperature rises. The pressure in the bottles is checked by pressure gages. The pressure must not exceed the values indicated in the operating instructions.

The extinguishers installed on board the aircraft are checked periodically for leakage by weighing them. The intervals for performing this operation are defined by the maintenance regulations. The permissible variation of the bottle charge weight depends on the extinguisher type, but usually does not exceed 100-200 grams. Newly recharged bottles are checked for leakage (after waiting for 24 hours) by applying soapy foam to the safety plug opening and the threaded connection of the pressure gage to the stop-valve housing. If no bubbles are noted, the extinguisher is considered tight.

At the intervals established by the regulations, the inspectors of the State Technical Inspectorate check the condition of the bottles and stamp a mark indicating the next required inspection date. To ensure operational reliability of the extinguisher, it is necessary to verify systematically the condition of the detectors and electrical circuits of the fire extinguishing system. Contact of water, fuel, oil, and hydraulic fluids with the cartridges and electrical wiring plugs is not permitted. During surveillance of the fire protection equipment it is necessary to check the condition of the distribution lines and spray manifolds. They should not have cracks, loose connections, or plugged orifices.

Care should be taken to ensure that the interior surfaces of the engine nacelles are clean, and accumulations of oil and fuel on these surfaces should not be permitted. At the intervals established by the maintenance regulations, and when changing engines, the vent lines are blown out with air, making certain that they have a continuous slope to the fuel recovery tanks.

The flight crew first learns of a fire by illumination of the warning panel (lights) and in certain aircraft types by means of an audio signal. When the fire warning comes on, the engine must be shut down, fuel shutoff valve closed, and air bleed into the air conditioning system must be closed off. On airplanes with PE and TPE the propeller is also feathered. Some time (usually 10-15 sec) after activation of the first bank of extinguishers, the warning system should be checked to see if the fire is out. If the fire is not out, the second extinguisher bank is activated.

It is recommended that the engine not be restarted after putting the fire out, since the fire cause and damage to the airframe and engine are not known. After use of the fire extinguishing system it must be put back into operating condition. To do this the bottles are removed from the airplane and charged, and the lines and manifolds through which extinguishers with the "3.5" agent have discharged are purged with carbon dioxide and checked for leaks. An engine cannot be approved for further flight after an engine fire. It must be removed from the airplane and sent to overhaul.

In case of extinguisher discharge when there is no fire in the powerplants and entry of the "3.5" agent into the engine, the latter may be approved for further operation if the engine oil is changed twice within a period of 1-2 hours after the agent gets into the engine. The temperature of the oil used for reservicing must be 50-70°C. The engine must be started up and run at less than rated power after each oil change. The reason for this is that the "3.5" agent lowers the oil viscosity and lubricating qualities. The engine is washed with warm water to remove traces of the liquid from the exterior surfaces.

Personnel involved with removing, installing, transporting, charging, and discharging the extinguishers must know and observe strictly the safety engineering rules concerning handling compressed gases and toxic fluids. We note that during discharge a large reactive force acts on a bottle charged with extinguishing agent at high pressure. Sudden self-opening of the stop-valve when the extinguisher is being held in the hands or intentional opening of the stop-valve to discharge the charge into the atmosphere without securing the bottle leads to severe accidents, since a man's strength is not sufficient to hold the discharging extinguisher.

Transportation of extinguishers in the unpacked form is accomplished on special carts, and they must be in a vertical position with the stop-valve up. Safety measures must be taken when working with extinguishing agents, since the decomposition products of these substances (for example, freon $114B_2$) are toxic and may cause poisoning.

CHAPTER VIII

INDUCTION SYSTEMS

Purpose and Requirements

The induction systems consist of the inlets (air intakes, diffusers), mechanisms for regulating the air flowrate, and equipment to protect the engine against foreign object entry.

The inlets are designed to supply the required amount of air to the engine. They may be an integral part of the engine or part of the airframe. These devices must provide the highest possible values of the total pressure recovery factor, low external drag, adequate uniformity of the flow at the compressor inlet, and stable and reliable engine operation in all flight regimes and engine operating conditions. The gas turbine engine pressure ratio π is defined by the equation

$$\pi = \frac{p_K}{p_H},$$

where p_c — is the compressor outside pressure;

p_H — is atmospheric pressure.

The pressure rise takes place in the inlet and in the compressor, therefore

where π_{in} — is the inlet pressure ratio;

π_c — is the compressor pressure ratio.

The efficiency of the air deceleration in the inlet is determined by the pressure losses in retarding the flow and by the losses owing to air friction with the inlet wall and ducts which direct the air to the engine.

The losses which arise during compression of the air in the inlet are evaluated by the total pressure recovery factor σ_{in}, which is the ratio of the total pressure p_1^* at the compressor inlet to the total pressure p_H^* of the adiabatically decelerated air (without losses):

$$\pi = \pi_{вх}\pi_{к},$$

Reduction of σ_{in} leads to decrease of the pressure ($\pi_{in} = \sigma_{in}\frac{p_H^*}{p_H}$) at the compressor inlet, decrease of the thrust and specific thrust, increase of the specific fuel consumption and powerplant weight. Thus, reduction of σ_{in} at M = 2.5 from 17 to 12, i.e., by 30%, leads to 45% reduction of the engine thrust and 15% increase of the specific fuel consumption. Therefore one of the important requirements imposed on the inlets is that they provide a supply of air with the highest possible value of σ_{in}.

Supply of the required amount of air is ensured by proper choice of the inlet area and variation of the inlet geometry as a function of the engine operating conditions and aircraft flight regimes.

Installation of an engine on a flight vehicle leads to increase of the frontal (external) drag Q_{ex}. This drag is created by the inlets, air intakes used for cooling the accessories, and by the engine nacelles. This includes nacelle and inlet wave drag, friction drag, and interference drag.

The magnitude of the external drag depends on engine installation configuration, inlet goemetry, and the condition of the wetted surfaces. Part of the thrust developed by the powerplant is expended in overcoming this drag. When the engines are located in the fuselage the external drag increases only slightly, however the internal hydraulic losses of the long ducts are large. On subsonic airplanes this reduces engine thrust and economy up to 15%. Placement of the engines in nacelles outside the fuselage increases the external drag, but the internal drag increases only slightly, which causes a loss of thrust and economy at subsonic flight speeds of no more than 2%.

The effective powerplant thrust P_{eff}, i.e., the thrust used to propel the flight vehicle, can be expressed by the formula

$$P_{эф} = P - Q_{вн} = P_0 - \Delta P_{вх} - Q_{вт},$$

where P_0 — is the engine thrust for $\sigma_{in} = 1$ (inlet without pressure losses);

ΔP_{in} — are the thrust losses caused by the pressure loss.

The inlet which makes it possible to obtain the maximal engine thrust P is termed <u>optimal</u>, while that providing the maximal effective thrust P_{eff} is termed <u>best</u>. For a flight vehicle we need to have the best inlet, for which ΔP_{in} and Q_{ex} are minimal.

The inlets of multi-regime flight vehicles must provide reliable and effective operation of the engines over a specified range of flight and engine operating regimes. Expansion of the inlet operating regimes reduces the effective thrust, since this involves the use of a complex inlet regulation system.

Air leads in the inlets of modern aircraft are unacceptable. Under the influence of the high pressure, air leaks through any cracks and creates additional drag. The air ducts are sealed by installing sealing strips in the joints, applying special sealants, pastes, cements, and lacquers.

It is particularly important to protect the inlet against entry of foreign objects during engine operation on the ground, since this can lead to engine failure.

Classification

Selection of the inlet depends on the airplane design flight speed, powerplant location, engine type, and other factors.

Depending on the flight speed, the inlets may be subsonic or supersonic. Depending on the engine installation configuration and inlet geometry, the inlets are divided into nose, side, and wing. The nose inlets are located in the nose of the fuselage or in the nose of the engine nacelles (Figure 149). The side inlets are located on the fuselage; they may be semicircular, two-dimensional, wedge-type, or scoop-type (Figure 150). The wing air intakes are located in the leading edge of the wing near the fuselage.

Subsonic Inlets

The subsonic inlets are made in the form of a duct with varying cross section area, in which the air stream is decelerated. This duct must be divergent. As the air travels along the duct

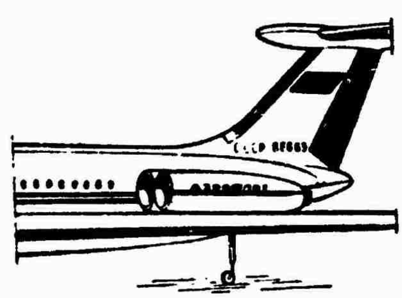

Figure 149. Nose air intake

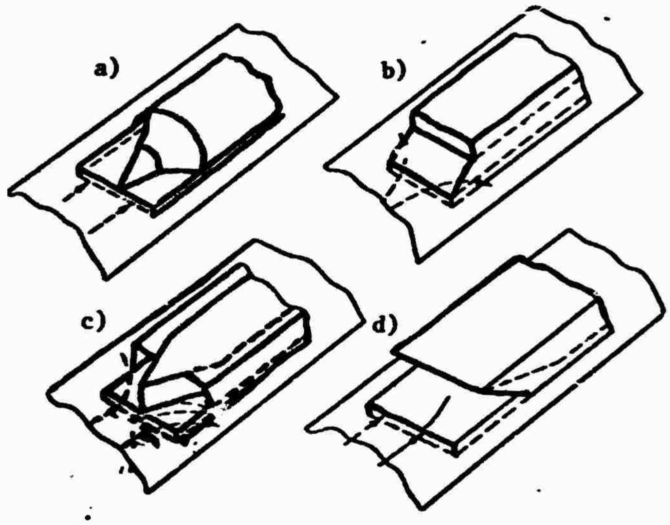

Figure 150: Inlet diffuser types:
a — semicircular; b — two-dimensional; c — wedge;
d — scoop

its velocity decreases and the pressure increases. The intensity of the deceleration is defined by the degree of variation of the inlet duct area: the larger the duct area variation, the greater the deceleration. However, the degree of flow deceleration is limited to prevent flow separation and reduce the inlet losses.

The inlet must have small hydraulic losses and low drag. These requirements are contradictory and selection of the optimal characteristics requires wind tunnel testing.

The diffuser profile is selected so that it will have smooth contours with large radius of curvature in the middle section and smooth increase of the curvature in the lengthwise direction. If the diffuser has rectilinear walls, its divergence halfangle (the angle formed by the axis of the air intake and its internal wall) must not be more than 4 - 6°. The entrance edges are smoothly rounded. The air flow ahead of the inlet depends on the relation between the flight velocity V and the velocity c_a, in the inlet.

During static engine operation (Figure 151a), when the flight velocity $V = 0$, the flow velocity ahead of the diffuser increases from zero at the undisturbed flow boundary to $c_{a'}$ at the compressor inlet. If $V = c_{a'}$, the air stream enters the diffuser without change of form and deceleration of the air takes place inside the diffuser (Figure 151b).

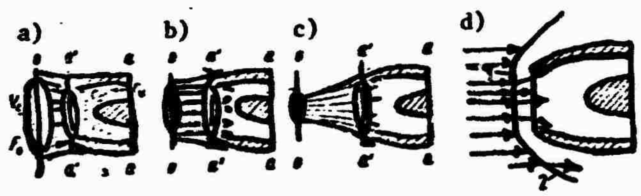

Figure 151. Flow in subsonic diffuser for subsonic and supersonic stream velocities:
a — subsonic flow is accelerated; b — subsonic flow remains unchanged; c — subsonic flow is decelerated; d — supersonic flow; 1 — normal shock; 2 — oblique shock

For $V > c_{a'}$ ($M < 1$) ram compression of the air (deceleration) begins outside the diffuser (Figure 151c) and terminates in the diffuser. Usually the air is decelerated so much prior to section a' - a' that its velocity increases from this section to the compressor face with corresponding pressure reduction, which leads to reduction of the hydraulic losses.

For subsonic air flows in the inlets, the pressure losses from ram compression are not large. They are caused basically by hydraulic resistance in the inlet ducts and $\sigma_{in} = 0.95 - 0.98$.

At the design flight Mach numbers the diffuser entrance area F_{in} is greater than the area F_0. With decrease of the speed the area F_0 increases and for some speeds becomes larger than F_{in}. Consequently, in subsonic diffusers the air compression in the design regime begins ahead of the diffuser. This reduces the diffuser hydraulic resistance, which is particularly important for long air inlet ducts. A high degree of external deceleration is also advisable for the side-mounted air intakes, which usually have curved inlet ducts.

For subsonic diffusers the inlet velocity c_a, is usually half the flight speed V, or even less. Maximal utilization of external deceleration of the approaching flow can be obtained under these conditions. When the air intake is short it is advantageous to have a large degree of external deceleration with subsequent acceleration to the velocity c_a. In this case the flow acceleration leads to equalization of the velocity field. In the case of long air intakes it is necessary to provide a high degree of deceleration to reduce hydraulic resistance.

At supersonic flight speeds a curved shock, termed the bow shock, is formed at some distance from the diffuser. Prior to air entry into the diffuser a normal shock is formed, whose surface is perpendicular to the flow direction. At larger distances from the inlet the shock becomes oblique and at a considerable distance from the diffuser transitions into a compression wave. The velocity becomes subsonic behind the normal shock and thereafter the flow remains the same as that discussed above. The higher the flight Mach number, the more intense the normal shock and the more energy there is converted into heat. The shocks reduce the total pressure of the air passing through them and the coefficient σ_{in} decreases. The pressure losses in the normal shock are evaluated by the total pressure recovery factor σ_t, which depends on the flight Mach number. In the presence of a shock the pressure losses are evaluated by the quantity σ_{in} = 0.95 - 0.98) σ_t.

The pressure recovery coefficient σ_m in a system of shocks (Figure 152) is the product of all the σ_i of all the shocks. We see from the figure that with increase of the Mach number the coefficient σ_m decreases rapidly and for M = 2.5 amounts to only 0.5 (curve 1). This means that the total and static pressure of the air behind the normal shock will be half of the values corresponding to adiabatic deceleration of supersonic flow (without losses).

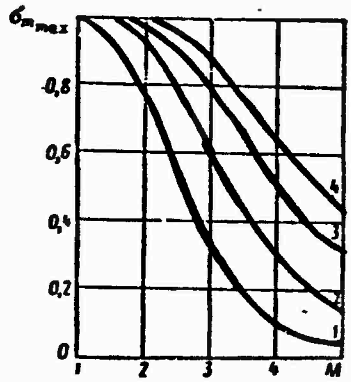

Figure 152. Coefficient $\sigma_{m_{max}}$ versus M for different shocks: 1 — normal shock; 2 — oblique shock + normal shock; 3 — two oblique shocks + normal shock; 4 — three oblique shocks + normal shock

For relatively low flight speeds (M ≤ 1.5) the pressure losses in the normal shock are comparatively small (σ_m = 0.92 - 0.93). Therefore for airplanes with these speeds we use the same inlets as for subsonic flight speeds.

The operating regime of the diffuser with a normal shock in the inlet may be subcritical, critical, or supercritical. In the subcritical regime there is a bow shock (Figure 153a), in the critical regime the normal shock is located in the entrance plane (Figure 153b), and in the supercritical regime it is located inside the diffuser (Figure 153c). Here the effective thrust is

$$Q_{эф} = P - Q_{вн} - Q_{доп}.$$

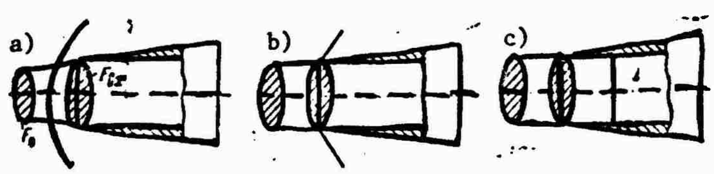

Figure 153. Diffuser operating regimes with normal shock at entrance

where Q_{add} — is the additional resistance arising from performing work in compressing the air which passes through the shock but does not enter the engine.

Obviously, for the critical and supercritical regimes Q_{add} is zero.

Although the losses in the normal shock at low Mach numbers are not large, they introduce disturbances into the airstream and cause boundary layer separation from the diffuser walls. To prevent this the diffuser leading edges are made sharp, with a small divergence angle, thereby avoiding abrupt turning of the flow, and an attempt is made to attach the shock itself at the design condition to the leading edge of the air intake.

Supersonic Inlets

In designing supersonic air intakes we attempt to obtain large values of the effective thrust at supersonic flight speeds. We see from Figure 152 that increase of the effective thrust can be obtained (by increasing σ_m) if the deceleration process is realized in a system of shocks rather than in a single shock.

The oblique shock is weaker than the normal shock and the pressure losses in it are less. If the Mach number behind the oblique shock $M \leq 1.5$, then the further deceleration is terminated by a normal shock, termed the closing shock. If there is a single oblique shock and a single normal shock, the diffuser is termed a double-shock diffuser. Such shocks are used for $M = 1.5 - 2$. With further increase of the flight Mach number ($M > 2$), still another oblique shock must be created behind the first oblique shock. Therefore, depending on the flight speed the diffusers may be double-shock, triple-shock, and quadruple shock. The closing shock is always normal.

The required system of shocks can be formed by means of a profiled centerbody which protrudes forward and has a stepped form. The shocks are obtained as a result of reflection of the waves from the inner cone. If there is no centerbody, a definite number of shocks can be obtained from a specially profiled inlet duct. A combined shock generation technique can also be used. Accordingly, supersonic diffusers are subdivided into three types: with external, internal, and mixed compression.

Depending on the location of the engine on the airplane, the supersonic diffusers with external compression are made in the form of an axisymmetric duct with sharp leading edges, inside which there is a central cone, or in the form of a two-dimensional duct formed by two asymmetric wedges, the larger of which plays the same role as the central cone (Figure 154). The lines Oa' denote the oblique shocks formed by the cone or upper wedge points, ca' are the shocks formed by the break in the cone, and a'b are formed by the point of the lower wedge. The slopes of the shocks to the diffuser axis are denoted by α. Behind these oblique shocks the air velocity is still supersonic. Further deceleration to subsonic velocity takes plane in a complex system of reflected shocks located in the engine duct, which are terminated by a single weak closing shock, denoted by the lines a'd and bd. The design regime of the inlet diffuser is that in which the first oblique shock is tangent to the cowling leading edge.

Since fixed geometry inlets operate well at supersonic speeds only at the design regime, their application is limited. The inlet diffusers have variable geometry in order to broaden the design regimes. These devices provide matching of the output of the shock system, the inlet throat (minimal section of the inlet duct), and the compressor. As a result of this matching we obtain maximal effective thrust and stable engine operation over a wide range of airplane flight speeds and engine operating regimes.

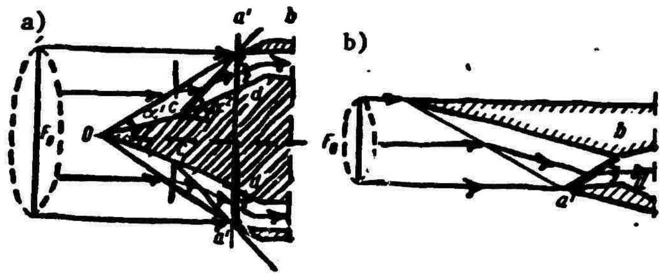

Figure 154. Supersonic external-compression diffusers with two oblique shocks:
a — with central cone; b — two-dimensional diffuser formed by two asymmetric wedges

Matching of the output of the inlet with the required air flowrate can be accomplished by:

varying the inlet area by deflecting the cowling edge (Figure 155a) or other elements of the air intake;

dividing the supersonic flow (Figure 155b);

use of a special system to bypass air from the intake into the atmosphere (Figure 155c);

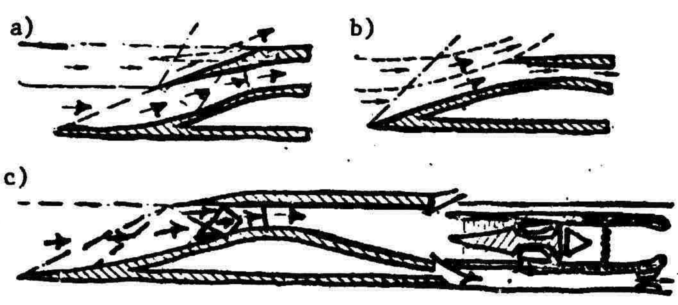

Figure 155. Techniques for providing required air flowrate through supersonic diffuser

admission of additional air into the diffuser duct from outside, bypassing the throat to increase the diffuser handling capacity and reduce the diffuser losses for Mach numbers M < 1 and during takeoff;

longitudinal displacement of the central cone to prevent detachment of the external oblique shocks from the cowling leading edge.

The inlets have quite varied geometry, depending on the engine type and its location on the flight vehicle, maximal flight speed and speed range, means for regulating the air flowrate, use of protective devices and so on.

The TPE inlet (Figure 156) consists of the propeller spinner 1, gearbox fairing 2, air intake 3, and engine inlet 4. The air intake and propeller spinner have profiled contours to ensure smooth air flow over them with small losses, without flow separation. They are fabricated from AMg and D16 sheet material. The skin is reinforced by internal members to stiffen all the inlet components.

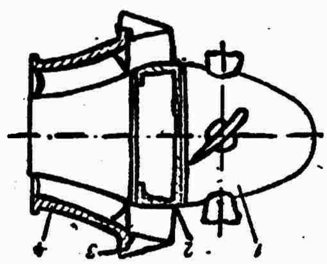

Figure 156. Turboprop engine inlet

A schematic of a side-mounted diffuser for a supersonic airplane is shown in Figure 157. The entrance and centerbody are semicircular and act like a circular diffuser which has been cut in half. A slot for boundary layer bleed is formed between the fuselage surface and the diffuser wall. The semicone is stepped and displaces axially in the direction of the diffuser axis. This makes it possible to alter the location of the oblique shocks and the throat area. In this diffuser the ducts supplying the air to the engine are short, and the nose of the fuselage is freed for the installation of various equipment, cargo, or crew.

Regulation of this air intake is accomplished by a two-step wedge (surfaces 1 and 2). The surface 1 is stationary and aligned in the approaching flow at a small angle (9°). The surface 2, which is the face of the second wedge, is hinged to the aft end of surface 1. The diffuser throat is formed by the movable surface 3 and the cowling. The elements 4 and 5 are hinged together and with the base of the inlet duct. Elements 2 and 3 are provided with special openings through which boundary layer suction is accomplished.

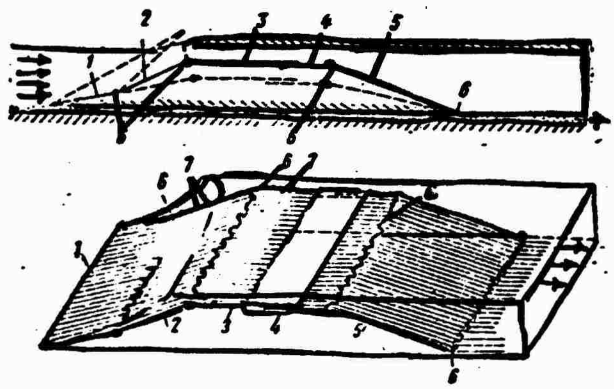

Figure 157. Two-dimensional supersonic controllable inlet with variable-geometry stepped wedge:

1 — stationary wedge; 2 — movable wedge; 3, 4, 5 — movable elements for regulating entrance area; 6 — hinges; 7 — boundary layer suction holes

The four engines of the Tu-144 airplane, which flies at M = 2.5, are located in a two-dimensional nacelle installed beneath the triangular wing (Figure 158). The two-dimensional inlet diffuser which supplies air to the engines is located in the forward part of the nacelle. A profiled wedge serves as the diffuser centerbody.

At high supersonic flight speeds, the powerplant and the airframe constitute an integrated whole. With increase of the flight Mach number, the powerplant specific weight increases continuously and the airframe specific weight decreases.

GRAPHICS NOT REPRODUCIBLE

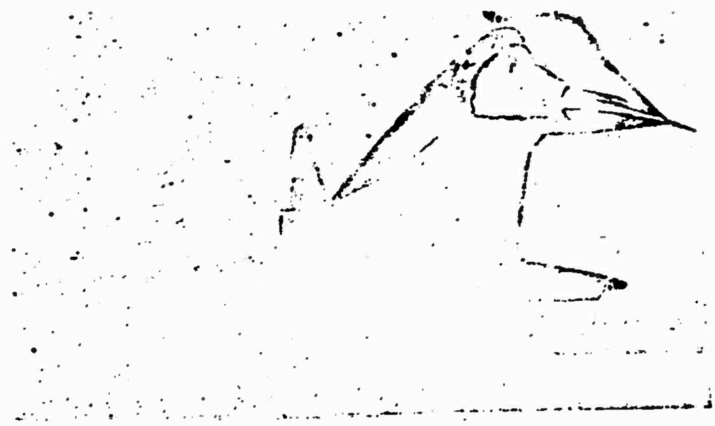

Figure 158. Tu-144 air intake

As an example, we shall consider a powerplant with an afterburner (Figure 159). This powerplant is designed for a supersonic passenger airplane. During the takeoff ground run (I) the inlet is designed so that the engine develops maximal thrust without afterburning. The second stage — takeoff climb (II), requires acceptable values of the noise level. Therefore the engine is throttled and the nozzle is set in the position to obtain minimal propulsive jet velocity. Position III provides maximal thrust for passage through the speed of sound. During flight at cruising speed (M = 2.2), the engine

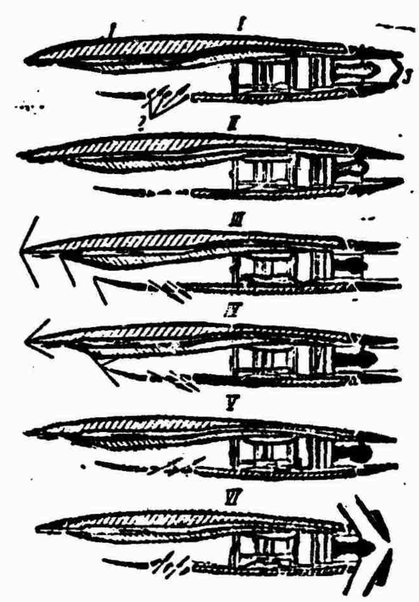

Figure 159. Supersonic inlets and exhausts for TJE:

1 — wing; 2 — controllable bypass louvers; 3 — controllable exhaust

operates with minimal fuel consumption and minimal external drag (IV). During transition to subsonic flight the powerplant is regulated to provide normal operation without afterburning (V). Position VI shows operation of the powerplant on the ground when using negative thrust for braking.

The supersonic inlets and exhausts must be designed as an integral part of the engine (powerplant). Engine development and testing must be done together with the induction and exhaust systems in order to determine the optimal engine operating regimes in the speed range for the given flight vehicle.

As a rule, the lift engines on VTOL aircraft are mounted vertically. The air intakes of these engines are made so as to increase the inlet pressure at low engine rotational speeds and ensure free entry of the air at high engine speeds, and they should also create a uniform velocity pattern at the engine inlet in the various VTOL flight regimes. In view of the large volume occupied by the air intakes and the air bleed system, the drag is increased.

The air intake of the VTOL engine includes inlet screens designed to stabilize the air flow at the entrance, upper louvers, and devices for improving engine operation. Exhaust gases may be drawn into the air intakes, and this leads to loss of thrust and compressor surging. Entry of these gases can be eliminated by deflecting the thrust vector, reducing the discharge velocity, screening, and using special louvers. When using a fanjet engine the temperature reduction in the air intake because of exhaust gas entrainment is reduced by using separate exhaust nozzles for the fan and the gas generator. The cold fan jet screens the hot jet from the gas generator. The design of the engine air intakes and their location on VTOL aircraft still require careful study.

Protecting the Engine Against Entry of Foreign Objects

Protective devices are installed in the inlet duct to prevent entry into the engine of dust, sand, rocks, pieces of concrete, and other objects which could cause damage to the engine parts or erosion of the surfaces of the compressor, combustion chambers, turbine, and can clog air orifices, bearings, and so on.

Entry of foreign objects happens most often during operation of the engine while parked, taxiing, and during takeoff and landing. Small objects which are sucked in reduce engine operating life, reduce thrust, increase specific fuel consumption, and in certain cases may cause engine failure. The higher the air flowrate and the closer the engine is located to the runway surface, the more probable it is that foreign objects will get into the engine.

Screens and grids which are either permanent or retractable after takeoff are installed in the inlet duct to protect the engine. Open protective screens (clearance about 4 mm) have been used to remove sand and dust from the air.

Icing can easily form on the protective screens. Therefore they must be heated and retractable in flight. Retraction of the protective screens can be accomplished by power cylinders activated by gear retraction and extension. The protective devices increase the engine specific weight and drag.

Attempts have been made to use an air curtain to protect the TJE against entry of foreign objects (Figure 160). For this purpose part of the air is bled from the engine and directed under pressure in the form of a transverse jet through special nozzles to intercept the vertical air flow coming from the ground. This scheme is usually activated by the gear: off when the gear retracts and on when it extends. However, the problem of reliable protection of the engine air intakes against the entry of foreign objects has not yet been entirely solved.

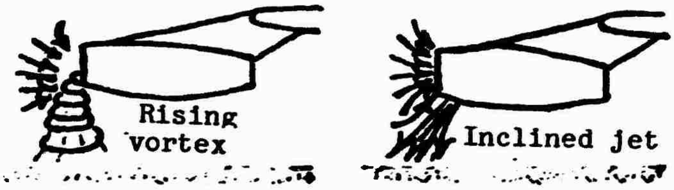

Figure 160. Operation of device to prevent sucking water, dust, and small objects from runway surface into TJE

Systems for Injecting Water into Air Intakes

The water injection system is designed to restore engine takeoff power or thrust at high air temperatures and low atmospheric pressure. When water is injected into the engine inlet duct, the air mass is increased and its temperature is lowered. Cooling of the air flow increases the flow total pressure, increases engine throughput and compression ratio. Thus, water injection increases the air and fuel flowrate into the engine and thereby maintains takeoff thrust or power.

When the water injection system is turned on (Figure 161), the valve 1 opens and air from the compressor enters the water tank 4, forcing water to the nozzles 14. Monitoring of system operation is

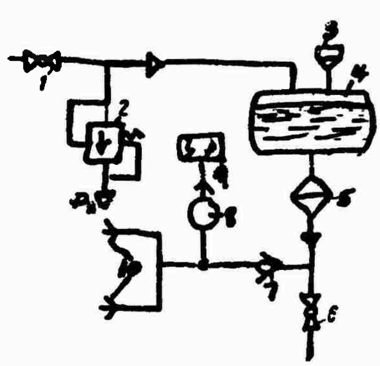

Figure 161. Schematic of system for injecting water into engine compressor inlet:

1, 6 — valves; 2 — relief valve; 3 — filler neck; 4 — water tank; 5 — screen; 7 — check valve; 8 — pressure switch; 9 — indicator light; 10 — nozzles provided by the signal light 9, which is energized by the pressure switch 8 when water is supplied to the manifold.

The water injection system is turned off after all the water is expended from the tank and the lines and nozzles are purged. Distilled water or water obtained by condensing steam is used for the injection system.

Maintenance

To ensure small air losses in the ducts, it is necessary to monitor the surface condition of the inlets, absence of dents, cracks, corrosion, and scratches. Particular attention should be devoted to sealing of the inlets. On today's supersonic airplanes air leaks through incidental cracks is completely unacceptable, since these leaks cause additional drag and reduce fire safety.

The elastic padding between movable components is covered with thermo-resistant insulation. The air intakes are covered with plugs when the airplane is parked to prevent entry of dust, moisture, sand, and other objects into the inlets.

CHAPTER IX

EXHAUST SYSTEMS

Purpose. Requirements. Classification

The exhaust systems are used to transform the potential energy of the gas flow into kinetic energy, discharge the gases into the atmosphere, and also protect the airframe components in the engine region against heating. These systems may be subsonic or supersonic, with controllable or fixed propulsive nozzle, with or without afterburning.

The primary part of the system is the outlet, consisting of the exhaust pipe and the propulsive nozzle. Some engine types include thrust reversal and noise suppression systems. If the engine is located in the forward part of the nacelle, or in the forward or middle part of the fuselage, a tailpipe is installed between the engine and the propulsive nozzle.

Structurally, the exhaust system consists of the following basic elements (Figure 162): exhaust pipe, inner cone, struts, tailpipe, propulsive nozzle, and thermal protection.

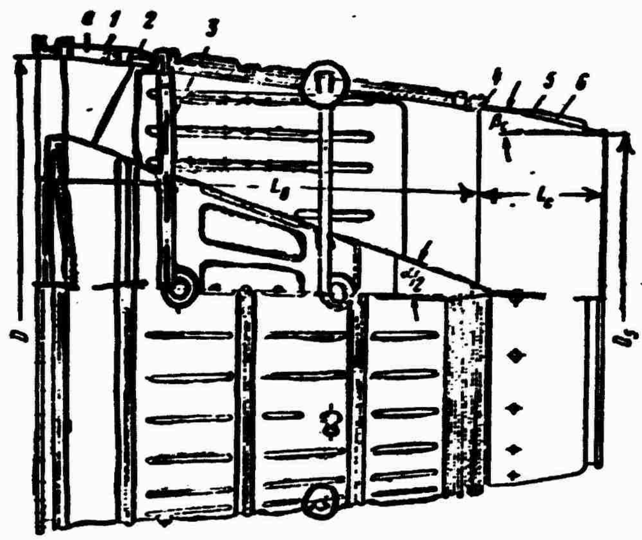

Figure 162. Exhaust system:

a — annular area; 1 — exhaust pipe; 2 — inner cone; 3 — rod; 4 — flange; 5 — shroud; 6 — nozzle

The exhaust pipe is attached by a flange to the turbine case and is most often made in the form of a truncated cone, which together with the inner cone forms a slightly divergent duct. The length of the exhaust pipe is as short as possible: usually $L_e = (0.9 - 1.3)$ D.

If there is a tailpipe in the exhaust system, its diameter must be selected so that the gas velocity in the tailpipe does not exceed 150 - 200 m/sec.

The inner cone prevents rapid expansion of the gases downstream of the turbine (smooth transition of the annular flow at the turbine exit into the full flow downstream of the inner cone). The inner cone apex angle α = 35 - 50°.

The struts connect the inner cone with the exhaust pipe and straighten the air flow which has been twisted in the turbine wheel. If the flow swirl downstream of the turbine is significant, the struts will have a twist and also their width and number are increased.

If the engine does not have an afterburner, and the flight Mach number does not exceed 1.5 - 1.7, the propulsive nozzle area decreases along its length. The propulsive nozzle diameter D_5 is determined on the basis of gasdynamic analysis, and the nozzle length $L_n = (0.2 - 0.4) D_5$. Small losses are obtained as the gases travel through the propulsive nozzle if the cone angle $\beta_n = 10 - 12°$.

The structural exhaust system elements operate under high temperature conditions and are washed by chemically active gases. The exhaust gas temperature reaches 700° C or more, and when afterburners are used — 1600 - 1800° C at a pressure of 0.2 - 0.25 MN/m^2. Therefore the exhaust system components are made from high-temperature steels EI-402, YaIT, EI-435 alloys, and so on. Thermal insulation of the tailpipe walls is provided with the aid of asbestos or a layer of air flowing over the pipe.

To reduce losses and obtain the optimal engine characteristics, it is necessary to have straight exhaust pipes, since curved pipes increase the exit thrust losses. Installation of tailpipes reduces the thrust and increases the weight and fuel consumption. For TPE it is considered that the thrust losses resulting from the installation of tailpipes amounts to 0.3% per tailpipe unit length (L/D = 1). The tailpipe section area is selected to be the same as the area of the exhaust pipe of the engine without a nozzle. The use of tailpipes complicates the airplane design. In this case it is necessary to install additional mounting fixtures and ensure that acceptable temperatures are maintained around the tailpipe.

The required temperature is maintained by means of thermal insulation of the hot surfaces and by cooling. The temperature of the structural elements around the tailpipe must be less than 140° C in all flight regimes and under any atmospheric conditions. At the same time the cooling must not lead to marked reduction of the exhaust gas temperature along the pipe, since this can lead to considerable thrust loss.

The greatest difficulties in cooling the tailpipe are encountered with the engine operating on the ground, when there is no relative air flow over the pipe, and in flight at supersonic speeds, when heating of the structure as a result of deceleration of the air flow is possible. To reduce the tailpipe temperature during static engine operation use is made of ejection or suction effects. In the first case the tailpipe shroud extends beyond the propulsive nozzle; in the second case the propulsive nozzle extends beyond the end of the tailpipe shroud (Figure 163).

The tailpipes must be straight and short in order to have low hydraulic resistance. The tailpipes of the TPE are an exception. Since the gas exhaust velocity from these engines is low, curved exhaust ducts may be used but they must be smoothly curved.

The attachment of the tailpipe to the exhaust pipe must be leakproof, provide free movement of the pipe during heating, and the possibility of some deflection of the tailpipe axis from the engine axis. The leak tightness eliminates exhaust gas and fuel leaks and is achieved by coating the joints with special enamels. Poor sealing can lead to fires. A fire hazard exists during restart attempts and when depreserving the engine. In these cases one or more motorings of the engine are performed to eliminate the fire danger. Drain provisions are made to drain off the fuel which may seep through the seal at the point where the tailpipe joins the exhaust pipe in case of an unsuccessful start.

The possibility of deviation of the tailpipe axis from the engine axis facilitates its installation in the airplane and unloads the tailpipe attachment flanges. The tailpipe wall temperature reaches 550 - 650° C or considerably higher when afterburning is used. At this temperature the elongation amounts to about 11 - 12 mm per meter. Therefore the aft tailpipe attachment must relieve the loads on the front attach flange during thermal expansion.

The tailpipe must be light in weight and this includes the tailpipe itself, the weight of the thermal insulation layer, the shroud for the thermal insulation and the tailpipe, and the weight of the stiffening and attaching elements. The pipe wall is made thin (to 1 mm) to reduce weight. However, even with this thickness the tailpipe weight is quite large. The tailpipe must be easily removable and convenient for inspection. This requirement is the result of the necessity for periodic examination of the turbine blades and the tailpipe itself.

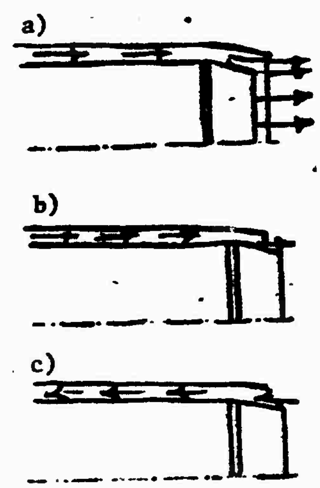

Figure 163. Air cooling of exhaust system with engine operating:

a — using ejection; b — in flight using ram pressure; c — on ground using suction

During takeoff and landing the exhaust jet, having high kinetic energy, may impinge on the runway and the adjacent surfaces of the flight vehicle. The jet velocity and temperature depend on the engine type; for example, for TJE the discharge jet velocity is 720 m/sec and the temperature is 820° C, while for TFE the corresponding figures are 500 m/sec and 350 - 500° C. Therefore the tailpipes must be located so that the discharging gases have little effect on the runway and do not impinge on the aircraft structure. However, if the jet does strike an aircraft surface the corresponding surface must be cooled and made of heat-resistant material.

The exhaust ducts can affect the stability and controllability characteristics because of the fact that the axes of the air inlet into the engine and the exit jet do not coincide, or because of the fact that the flow pattern over the empennage changes because of the action of the discharging gases. Moreover, the exhaust system should not affect the engine characteristics during start.

The need for the installation of a tailpipe and its dimensions are determined by the engine positioning on the aircraft. The exhaust system with short tailpipe is simplest (Figure 164). It consists of a tailpipe made from 1Cr18Ni9A steel, a shroud, and the inner cone. To increase the strength and stiffness the tailpipe has stiffening bands, and the shroud has stiffening bands and ribs. The pipe is made from 1-mm-thick sheet steel by stamping and roll welding. The bands and ribs are spaced at intervals of 400 - 500 mm. A flange is welded to the pipe for attachment to the engine.

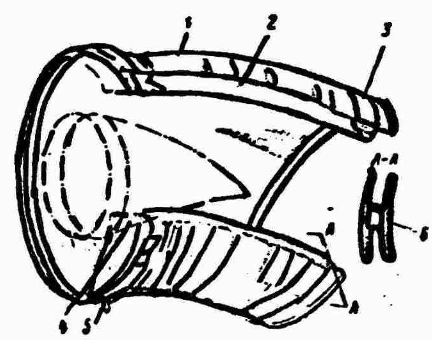

Figure 164. TPE exhaust system with short tailpipe:
1 — tailpipe shroud; 2 — tailpipe; 3 — band; 4 — exhaust pipe; 5 — drain fitting; 6 — bracket.

Discharge of the gases through long tailpipes presents considerable difficulty (Figure 165). The exhaust system consists of the flange 1 attached to the engine, inner cone 2, tailpipe 4, and attaching parts. The aft attach fittings permit the pipe to move as it elongates. Since the pipe is quite long (about 5 m), special rollers are installed on the airplane and travel along guide profiles. This makes it possible to install and remove the tailpipe without applying large forces.

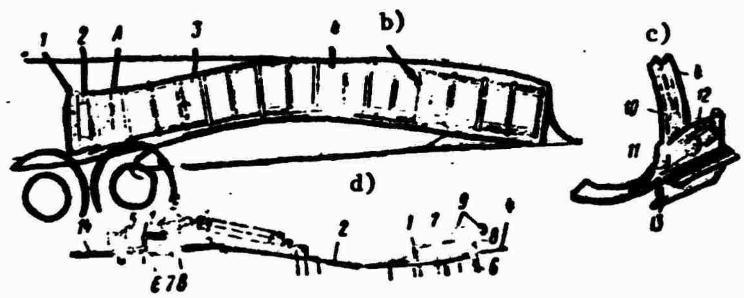

Figure 165. TJE exhaust system with long tailpipe:
1 — flange; 2 — inner cone; 3 — roller for removing tailpipe; 4 — tailpipe; 5 — bolt; 6 — reflector; 7 — ring with collar; 8 — joint clamp; 9 — joint bolt sleeves; 10 — bracket; 11 — nacelle rails; 12 — roller; 13 — nacelle; 14 - engine

A, B — front and rear attachments to nacelle; C — aft exhaust pipe attachment; D — connection of engine with exhaust pipe

A shroud which forms an annular duct around the tailpipe is installed in order to control the airflow which cools the pipe. Spacer brackets are riveted at the aft end of the shroud and bear on the tailpipe to ensure a uniform gap between the tailpipe and shroud.

Sometimes the tailpipe is covered with a thermal insulation layer (for the TJE and FJE exhaust systems). Matte foil, glass wool, and asbestos are used for this layer. The layer thickness may be 6 - 12 mm or more and a thermal insulation shroud is installed outside this layer. If the calculated thickness of the layer is 3 mm or less, the tailpipe is usually cooled by air.

The engine exhaust systems for supersonic airplanes are more complex structures. With increase of the flight speed there is an increase of the compression ratio so that, other conditions being the same, the turbine outlet total pressure p_2^* increases. The pressure ratio $\frac{p_2^*}{p_H}$ also increases, and at high flight speeds the pressure ratio in the propulsive nozzle becomes supercritical. If complete expansion of the gas to the pressure $p_e = p_H$ takes place in a simple nozzle, the specific thrust losses become noticeable at flight speeds corresponding to M = 1.5 - 1.6 and increase rapidly with further increase of M.

Therefore, for TJE intended for supersonic airplanes use must be made not of the simple expanding nozzles but rather supersonic nozzles, which for supercritical pressure ratios and a given value of the turbine-out gas temperature increase the specific thrust and reduce the fuel consumption. For example, at M = 2 and a flight altitude of H = 11,000 m the specific thrust of the TJE with a supersonic nozzle is 20 - 25% greater than for the same engine with a simple nozzle at the same conditions. This improvement becomes even greater at higher Mach numbers.

The length l_n of the expanding part of the supersonic propulsive nozzle is determined by the magnitude of the ratio $\frac{F_e}{F_{cr}} = \frac{D_e^2}{D_{cr}^2}$ and the divergence angle α_n (Figure 166). With reduction of the angle α_n for a given ratio $\frac{F_e}{F_{cr}}$ the length and area of the nozzle walls, and therefore the weight, increase; the losses and quantity of air required for cooling the walls also increase.

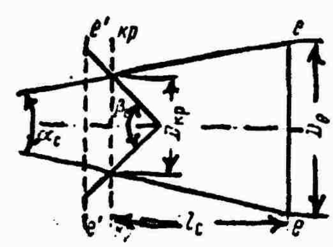

Figure 166. Supersonic propulsive nozzle

For excessively large divergence angles the flow separates from the walls and nonparallel gas flow develops, which leads to decrease of the thrust. Therefore, usually the angle $\alpha_n = 25 - 30°$ and the subsonic part of the nozzle is made with the contraction angle $\beta_n = 90 - 120°$.

The ratio $\frac{F_e}{F_{cr}}$ is termed the area ratio of the supersonic nozzle and depends on the degree of expansion of the gas in the nozzle $\varepsilon_{ne} = \frac{p_2^*}{p_H}$. With increase of ε_{ne} the required area ratio increases. When the area ratio provides complete expansion of the gas to atmospheric pressure, i.e., $p_e = p_H$ and $\varepsilon_{ne} = \frac{p_2^*}{p_H}$, this nozzle operating condition is termed the design regime. If the area ratio is not sufficient for complete expansion of the gas, the nozzle operates in the underexpansion regime $p_e > p_H$ while $\varepsilon_{ne} < \frac{p_2^*}{p_H}$. In the first case the engine thrust

$$P = \frac{G_s}{g}(c_e - V);$$

in the second case

$$P = \frac{G_s}{g}c_e' + F_e(p_e - p_H).$$

If the area ratio is greater than that necessary for expansion of the gas to atmospheric pressure, the nozzle operates in the overexpansion regime, i.e., at its exit section $p_e < p_H$ and $\varepsilon_{ne} > \frac{p_2^*}{p_H}$.

Since in this case the velocity at the nozzle exit is supersonic, the gas flow is decelerated with the formation of oblique shocks.

The flow velocity is determined by the actual expansion ratio $\varepsilon_{ne} = \frac{p_2^*}{p_e}$, corresponding to the given nozzle area ratio, and the engine thrust

$$P = \frac{G_s}{g}(c_e' - V) - F_e(p_H - p_e).$$

In the case of large overexpansion the oblique shocks propagate into the nozzle and separation of the jet from the nozzle walls is observed.

The pressure on the inner side of the nozzle surface aft of the separation plane is close to p_H. Therefore the part of the nozzle aft of the separation plane actually does not do any work (does not participate in creating the thrust force). For this case the thrust

$$P = \frac{G_s}{g}(c_x - V) - F_x(p_H - p_x),$$

where c_x, F_x and P_x — are the velocity, nozzle area, and pressure at the point when the flow separates from the walls.

At high supersonic flight speeds it is better to use a supersonic propulsive nozzle with small area ratio, which does not provide complete full expansion at the design flight altitude and speed. This decreases the nozzle length and weight significantly, and the diameter as well, which reduces powerplant drag markedly with only a small thrust decrease.

When the flight speed or altitude is changed the value of $\frac{p_2^*}{p_H}$ changes over wide limits. Thus, for the TJE with $\pi_K^* = 8$ and combustion chamber exit temperature 1300° K at the flight speed corresponding to M = 3 and an altitude of 20,000 m the ratio $\frac{p_2^*}{p_H} \approx 20$, while for this engine at sea level at maximal thrust the ratio is 2.9. Therefore, if the nozzle is designed to obtain complete expansion at the design flight altitude and speed, when operating at sea level it will operate in the large overexpansion regime, which causes large thrust losses.

Normal stable nozzle operation in all flight regimes is achieved by using a controllable nozzle, which provides complete or nearly complete expansion of the gas with minimal losses. To achieve this the nozzle divergence must be regulated in accordance with the variation of the engine operating conditions.

The variation of the exit area F_e and throat area F_{cr} of the supersonic propulsive nozzle can be obtained by displacing an internal spike (Figure 167a), rotating eyelids which form the nozzle walls (Figure 167c), displacing the center spike and rotating the flaps (Figure 167b), rotating the flaps and injecting air to alter the throat section (Figure 167d), and by other means.

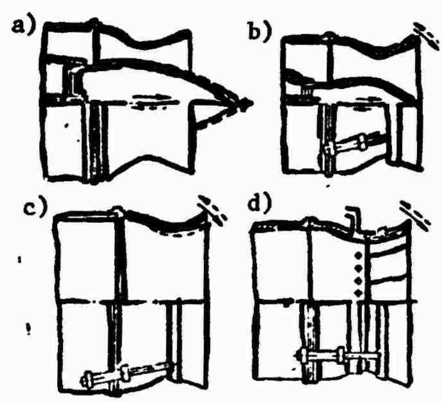

Figure 167. Controllable propulsive nozzles

The use of controllable nozzles increases the weight, complicates the structure, and requires the use of control systems. Quite often an afterburner chamber is installed in the exhaust system to burn additional fuel and thus increase the enthalpy of the air, increase the discharge velocity and engine thrust.

The basic elements of this chamber are the diffuser, flameholder, combustion chamber itself, equipment for injecting and igniting the fuel, and the propulsive nozzle.

Organization of stable combustion in the afterburner chamber is possible only if the air velocity at the entrance to the chamber is reduced to 120 - 200 m/sec. The diffuser is used to accomplish this. The flameholder, which creates zones of reverse air flow to ensure stable combustion, is located in the diffuser. Centrifugal-type injector nozzles, spaced uniformly along the perimeter of the flameholders and facing the incoming stream, are installed ahead of the flameholders on special fuel manifolds. The number of injectors is large (up to 200).

Reduction of the flow velocity by means of only a single diffuser is not advisable, since the dimensions of the diffuser itself increase. The afterburner chamber has a controllable nozzle (subsonic or supersonic). The nozzle eyelids open when the afterburner is activated and close when it is shut off. Opening must be quite fast to prevent increase of the turbine-out gas temperature and overheating of the turbine blades. It is recommended that closing be slower. Control of the eyelids and fuel flowrate must be coordinated in order to shut off the fuel flow if the eyelid control system fails.

All the afterburner chamber components are made from refractory materials. The outer surfaces are air cooled and the inner surfaces are coated with a special enamel.

The VTOL exhaust systems are quite varied and depend on the airplane type, the engines used, and the stabilization and control system.

With regard to techniques for creating thrust, the VTOL airplanes with a single powerplant are divided into two groups: tilting engine and exhaust jet deflection. The exhaust systems of the VTOL airplanes with tilting engines are similar to those discussed above. In the VTOL airplane with exhaust jet deflection there are special devices for rotating the gases to obtain vertical and horizontal thrust in a single engine. The exhaust system has two nozzles:

the primary nozzle and the nozzle for creating vertical thrust, and also a special diverter valve. If the diverter is positioned so that all the gas exhausts into the primary nozzle, only horizontal thrust is created (Figure 168a). The diverter can direct the gas through the nozzle to create vertical thrust. The vertical nozzle can deflect from the initial position, which provides for airplane acceleration after takeoff and deceleration prior to landing. This nozzle does not provide smooth thrust variation in the intermediate regimes.

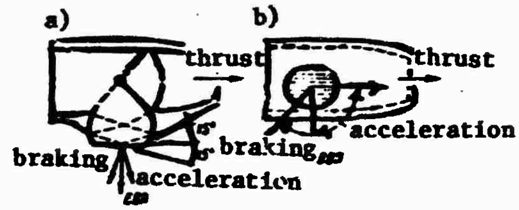

Figure 168. VTOL exhaust with jet deflection:

a — two position nozzle; b — all-regime nozzle

The all-regime nozzle, which is aligned horizontally (Figure 168b), eliminates this drawback. It has pivoting louvers and, depending on the louver position, can develop vertical thrust and provide acceleration or deceleration.

The VTOL airplane with lift-cruise engines also has devices for deflecting the jet. The lift engines are mounted at right angles to the airplane longitudinal axis. At the inlet to and exit from these engines there are louvers, which form the side walls of the air intake and the exhaust system. In horizontal flight these louvers are closed.

Turbofan units or gas ejectros can be used as thrust augmentors. In the exhaust system of such powerplants there is a diverter value to direct the gas to the turbofan unit or the ejector. The turbofan unit and the ejector have upper and lower louvers which provide for air intake and exhaust discharge.

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Thrust Reversal

The modern airplanes require long runways for takeoff and landing. The kinetic energy is absorbed in landing by aerodynamic forces, wheel friction on the runway, and braking parachutes. A considerable reduction of the ground roll distance can be achieved by using reverse thrust, which also improves airplane maneuverability. During thrust reversal the direction of the gas stream leaving the engine is changed by an angle α which varies from 90 to 180°.

The device used to rotate the thrust vector is termed a reverser. Depending on the operating regime of the TJE with reverser, the thrust may be positive, zero, or negative. The ratio of the thrust with reverser operating to the thrust with reverser off is termed the thrust reversal ratio

$$\varepsilon_p = \frac{P_p}{P}.$$

The following requirements are imposed on the reversers:

the magnitude of the negative thrust with reverser on must be at least 35 - 40% of the static thrust developed by the engine;

the thrust change must take place in a minimal time period (1 - 2 sec);

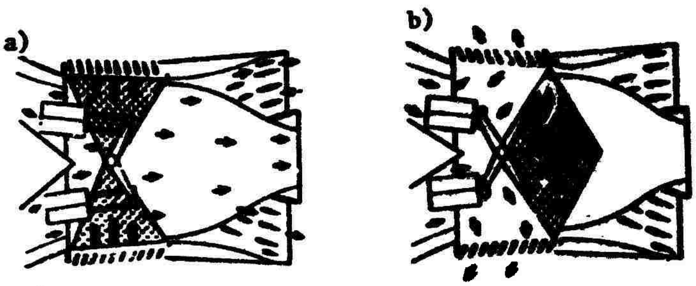
no change in engine operating conditions when the reverser is actuated;

simplicity and reliability of reverser construction, small size and low weight;

the exhaust jet must not heat the airplane skin;

a)

b)



c)

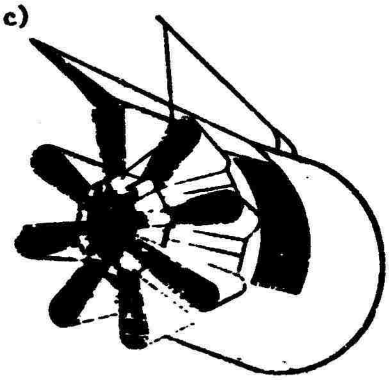


Figure 169. Mechanical thrust reverser with noise suppressor:

a — thrust reverser schematic; b — reverser-noise suppressor

absence of thrust asymmetry when activating the reverser, even when the airplane has several engines;

safety and reliability in operation.

Depending on the technique for creating the negative thrust, reversers can be divided into two types: mechanical and aerodynamic. In the first type (Figure 169) the reverser shutters rotate the gas stream, thus creating a negative thrust with reversal ratio 0.4 - 0.5. In the inoperative position the positive thrust losses do not exceed 2 - 3%. These devices integrate into the engine well and are simple in construction. A schematic of a reverser of the second type is shown in Figure 170. This unit is equipped with vanes which are aligned with the exhaust flow when the reverser is inactive. When the reverser is activated the vanes rotate and deflect the gas flow.

Thrust deviation is used on certain types of airplanes. A change of the thrust direction by rotating the thrust vector 90° toward the runway is termed <u>deviation</u>. This deflection of the thrust gives rise to a vertical thrust component, which balances the weight force and reduces the landing speed and landing roll. Deviation may be obtained by rotating the exhaust nozzle.

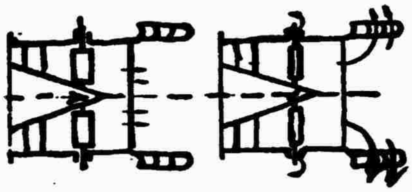


Figure 170. Aerodynamic thrust reverser

Noise Suppression

The noise created by aircraft has a harmful effect on both the aircraft itself and the surrounding medium. In airplanes with subsonic flight speeds the noise arises from engine operation, and on the TPE from the air propeller. When flight speeds exceeding the speed of sound were achieved, the sonic boom problem developed, the intensity of the boom depending on flight altitude and airplane size. During engine operation about 1% of the engine power is converted into noise energy, which is sufficient to create a sound (noise) pressure level above 170 dB. For supersonic passenger airplanes the noise level reaches 175 dB. It has been established that the noise level at the boundary of the airfield should be no more than 102 dB at night and 110 dB in the daytime.

The exhaust jet has the highest noise intensity. The noise of the compressor, turbine, and boundary layer is somewhat lower. The fan noise of the FJE may reach large magnitudes.

The noise generated by the engine can cause acoustic damage to the structure, and have a harmful effect on servicing personnel, passengers, and flight crew.

The exhaust jet noise level intensity depends on the jet density ρ_j, velocity V_j, and area S_j. For constant ambient conditions the jet density changes very little; therefore its influence on the noise intensity can be neglected. The primary cause for the appearance of noise is the jet velocity and its area. Noise reduction can be achieved by reducing the exhaust jet area or its discharge velocity.

Noise suppressors are used to reduce the noise. They reduce the noise level by splitting up the gas jet, deforming its cross section to the maximal degree possible, or dividing the jet up into small parts (Figure 171).

The use of noise suppressors leads to engine thrust losses. Thus, when reducing the noise 6 dB by a lobed suppressor (see Figure 168b) the thrust decreases by 0.4%, and for a 10 dB reduction the thrust drops by 1%. The optimal situation for this type of suppressor is division of the jet into six or eight sectors. Moreover, the suppressors increase the weight, fuel consumption, and drag.

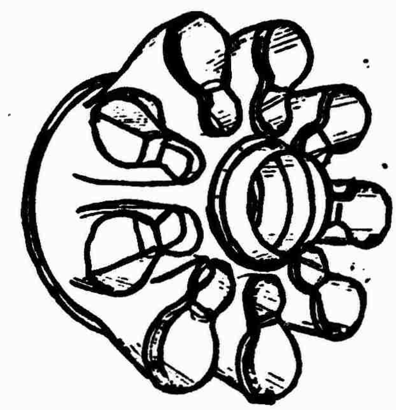


Figure 171. Noise suppressor with individual nozzles

Reduction of the noise level in the airfield environs can also be accomplished by use of the proper takeoff technique, i.e., by changing the thrust and flight profile.

Supersonic airplanes create considerable noise in the airfield region and cause the appearance of a sonic boom when transitioning from subsonic to supersonic speed. The sonic boom not only creates an unpleasant sensation, but at a definite intensity it may lead to structural damage. The boom intensity depends on the airplane flight weight and altitude (Figure 172).

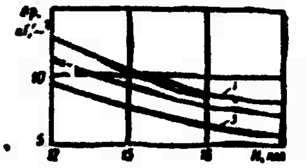


Figure 172. Sonic boom intensity on ground versus flight altitude and airplane flight weight:

1 — G = 180,000 kgf; 2 — G = 136,000 kgf; 3 — G = 91,000 kgf

Maintenance

Maintenance of the exhaust system during operation is accomplished in accordance with the maintenance instructions for the given airplane. The condition of the structural elements is checked, special attention being devoted to inspection of the exhaust pipe: checking for damage, warping, burnthrough, cracks; security of the attachment is verified and the clearance between the exhaust pipe and its shroud is checked. Lack of clearance leads to burnthrough and the appearance of cracks on the tailpipe and other components. During long-term engine storage the exhaust system is given both internal and external preservation treatment. Internal preservation is accomplished using engine oil; external preservation is provided by a special lubricant, and the attachment fittings are coated with grease.

CHAPTER 10

AIR PROPELLERS

Basic Characteristics

The air propeller is an assembly used to create a thrust force, which is the reaction of the air stream ejected by the propeller. In creating the thrust force the propeller converts engine mechanical energy into the work performed in the translational motion of the flight vehicle.

The following basic requirements are imposed on air propellers:

high efficiency;

automatic change of blade pitch as a function of airplane flight conditions and engine operating regime;

blade pitch range must provide for easy engine starting, minimal positive thrust at idle power, operation of the propeller in the negative thrust regime during landing roll, and feathering;

blade rotation rate when increasing pitch must be at least 10 deg/sec;

minimal values of the reactive and gyroscopic moments;

the design of the propeller and governor must include automatic protective devices to prevent negative thrust which operate reliably over the entire range of flight conditions and prevent unintentional movement of the propeller blades to low incidence angles;

protect the blades and spinner against icing.

Propeller Pitch, Advance, and Blade Incidence Angle

The propeller geometric pitch H is the distance a propeller would advance along the axis of rotation in a single revolution when screwing into a nut specially made for it, i.e.,

$$H = 2\pi r \operatorname{tg} \varphi, \tag{106}$$

where r is the distance from the blade rotation axis to the section in question;

ϕ is the blade section incidence angle.

Blades can be characterized by the relative geometric pitch

$$h = \frac{H}{D} = \pi \bar{r} \operatorname{tg} \varphi,$$

where D is the propeller diameter;

$\bar{r} = \frac{r}{R}$ is the relative radius;

R is the propeller radius.

We see from (106) that the pitch of a propeller of given diameter is completely defined by the magnitude of the incidence angle. In spite of the fact that in aircraft operation the value of the propeller pitch is never measured and only the value of the incidence angle is used, the term "propeller pitch" is very widely used. Since the propeller actually operates in air, which is an elastic, compressible and therefore, yielding medium, in one revolution the propeller will advance a distance which is considerably less than the geometric pitch H. This quantity is termed the propeller advance and is denoted by the letter H_a.

Clearly,

$$H_a = \frac{V_0}{n},$$

where V_0 is the airplane flight speed, m/sec;

n is the propeller rotational speed, rps.

In propeller calculations and design it is more convenient to use the propeller advance ratio $\lambda = \frac{H_s}{D}$. After multiplying the numerator and denominator by n we obtain

$$\lambda = \frac{V_0}{nD}.$$

The advance ratio λ is a dimensionless quantity. It is sometimes called the propeller regime characteristic or speed coefficient.

Blade Section Velocity Polygon

In flight any blade section rotates with the circumferential velocity U_0 and travels translationally with the velocity V_0 (Figure 173). In addition to these basic velocities, in the plane of rotation there arise the induced inflow velocity v_1 (in the axial direction) and induced swirl velocity u_1 (perpendicular to the propeller rotation axis).

The resultant velocity of the given blade section is

$$W_1 = \sqrt{(V_0+v_1)^2+(U_0-u_1)^2}.$$

The total induced velocity at the given propeller radius is

$$w_1 = \sqrt{v_1^2+u_1^2}.$$

The direction W_1 forms with the propeller profile chord the angle of attack α and with the direction of the circumferential velocity U the angle β, which is termed the stream inflow angle. Therefore the blade section incidence angle

$$\phi = \alpha + \beta$$

Figure 173 also shows the basic velocity triangle without account for the induced velocities v_1 and u_1. We see from the velocity diagram that the angle β differs from the angle β_0, included between the velocity W_0 and the circumferential velocity U_0, by the flow angularity $\xi = \beta - \beta_0$.

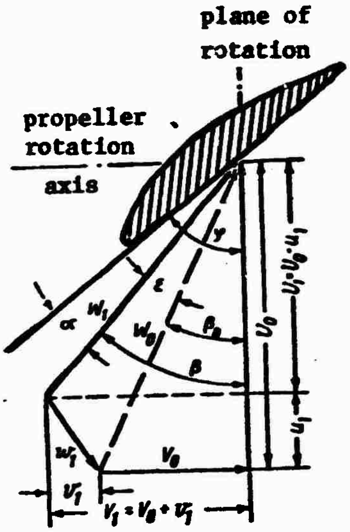


Figure 173. Blade section velocity polygon.

Blade Twist

In the early days, prior to study of the complex picture of propeller operation in a yielding medium, the back face of the blade was given the form of a geometric screw surface. These were propellers with radially constant pitch. It was later found to be more efficient to give all the blade sections the same optimal angle of attack α. For this the blade must have incidence angles which vary along the radius, or <u>twist</u>, so that the incidence angles are greater at the blade hub and decrease toward the blade tip. The twist should provide the condition

$$\alpha = \phi - \beta = \text{const} = \alpha_{opt}.$$

To define the magnitude of the blade twist we use the concept of blade section <u>relative twist</u> $\bar{\phi}$ (Figure 174), comparing the incidence angle ϕ of any blade section with the incidence angle of the section located at the radius $\bar{r} = 0.75$ and denoted in the form $\phi_{0.75}$:

$$\bar{\phi} = \phi - \phi_{0.75}.$$

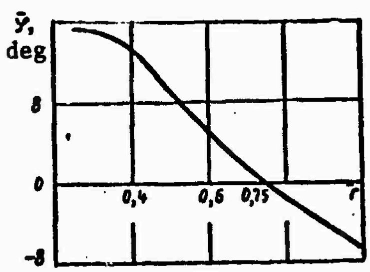


Figure 174. Change of relative blade twist angle along propeller radius.

The overall blade twist is determined by the difference between the incidence angle ϕ_{r_0} at the beginning of the working part of the blade and the incidence angle ϕ_R at the blade tip.

At the present time only propellers with pitch (incidence angle) which varies along the blade radius are used. The concepts of "propeller pitch" and "blade incidence angle" are also applicable for these propellers. However, for the variable pitch propellers these concepts apply to a single definite reference blade section at some nominal radius. For propellers of small diameter (to 3.5-4 m) this section is taken at a radius r_{nom} equal to 1000 mm from the axis of rotation. For propellers of larger diameter (more than 4.0 m) the reference sections are taken at a radius r_{nom} equal to 1600 mm from the propeller axis of rotation. These sections are marked on the blades by a narrow red paint stripe.

Thrust and Power

A propeller blade element bounded by two sections at the radii r and r + dr (Figure 175) is acted upon by the aerodynamic force

$$dR = c_R \frac{\rho W_1^2}{2} b\,dr,$$

where c_R is a coefficient which depends on the blade section profile and angle of attack;

ρ is the air density;

b is the blade element width.

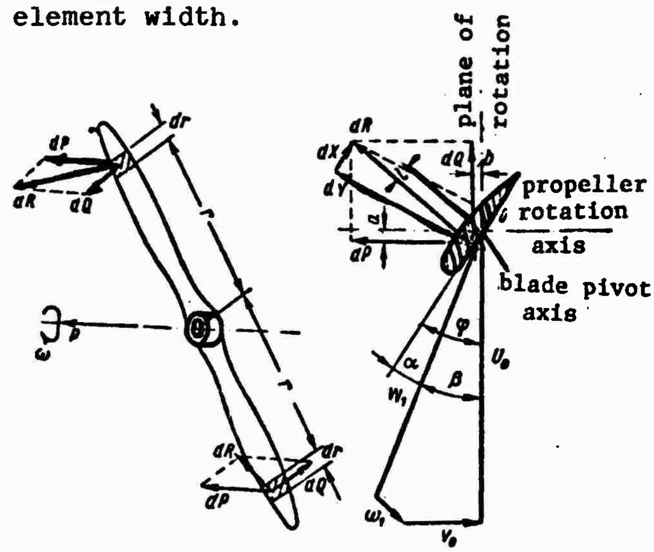


Figure 175. Aerodynamic forces acting on propeller blade element.

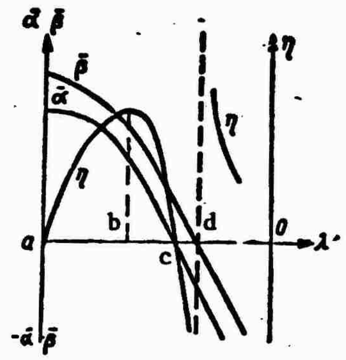


Figure 176. Aerodynamic characteristics of air propeller at constant pitch angle.

The projections of the aerodynamic force dR on the axis and plane of rotation respectively are the thrust dP of the element in question and the rotational resistance force dQ. The product of dQ by the element radius r yields the elementary torque dM = dQr, and after multiplying the latter by the angular velocity ω we obtain the power expended to rotate the element

$$dT = dQr\omega = dM\omega$$

If we integrate these expressions from the beginning of the working part of the blade ($r = r_0$) to the blade tip ($r = R$), we obtain the thrust, torque, and power required of the entire propeller with i blades

$$P = i\int_{r_0}^{R} dP \text{ N};$$
$$M = i\int_{r_0}^{R} dQr \text{ N}\cdot\text{m};$$
$$T = i\int_{r_0}^{R} dQr\omega \text{ W}.$$

The useful power T_p delivered by the propeller to propel the airplane is equal to the product of the thrust and the flight velocity: $T_p = PV_0$. The propeller efficiency is equal to the ratio of its useful and required powers

$$\eta = \frac{PV_0}{T}.$$

In accordance with aerodynamic similarity theory, the propeller thrust and power required have the following expressions

$$\left.\begin{aligned} P &= \bar{\alpha}\rho n^2 D^4; \\ T &= \bar{\beta}\rho n^3 D^5, \end{aligned}\right\} \qquad (107)$$

where $\bar{\alpha}$ is the dimensionless propeller thrust coefficient;

$\bar{\beta}$ is the dimensionless propeller power coefficient.

The coefficients

$$\bar{\alpha}=\frac{P}{\rho n^2 D^4} \text{ and } \bar{\beta}=\frac{T}{\rho n^3 D^5}$$

characterize the thrust and power required of geometrically similar propellers.

Using the last formulas for propeller thrust and power, we can write the efficiency in a different form

$$\eta=\frac{PV_0}{T}=\frac{\bar{\alpha}\rho n^2 D^4 V_0}{\bar{\beta}\rho n^3 D^5}=\frac{\bar{\alpha}V_0}{\bar{\beta}nD}=\frac{\bar{\alpha}}{\bar{\beta}}\lambda.$$

The coefficients $\bar{\alpha}$, β and λ (Figure 176) play an important role in all propeller calculations.

Operating Regimes

For a constant setting ϕ the blade angle of attack α depends on the flight speed (see Figure 173). The angle of attack decreases as the flight speed increases. In this case we say that the propeller "unloads," since the propeller rotational resistance torque decreases, and therefore the engine power required decreases. This causes an increase of the rotational speed. Conversely, when the flight speed decreases the angle of attack increases, the propeller "loads up," and the rotational speed decreases.

If there is a large increase of the flight speed or the incidence angle is small, the angle of attack may become zero or even negative. In the case $\alpha < 0$ the blades encounter the airstream with their forward face rather than with their aft or working face. In this case the thrust and power may become negative.

The thrust P and the thrust coefficient $\bar{\alpha}$ are considered positive if the thrust direction coincides with the direction of aircraft motion, and are considered negative when the directions are opposite. In the latter case the propeller creates drag.

The propeller power T and the power coefficient $\bar{\beta}$ are considered positive when the torque from the propeller aerodynamic forces is opposite the propeller direction of rotation. If the torque of these forces supports the propeller rotation, i.e., if the rotational resistance force $Q < 0$, the propeller power is considered to be negative.

When V_0 and n vary over a wide range the advance ratio λ may vary from zero to infinitely large positive values (as $n \to 0$). We shall examine the most characteristic propeller operating regimes.

The regime in which the translation velocity $V_0 = 0$, and therefore λ and η equal zero, is termed the <u>propeller static operating regime</u> (Figure 177). In Figure 176 this regime corresponds to point a, where the thrust $\bar{\alpha}$ and power $\bar{\beta}$ coefficients usually have their maximal values. The blade angle of attack α is approximately equal to the blade setting ϕ in the propeller static operating regime. Since $\eta = 0$, the propeller does not perform any useful work during static operation.

The propeller operating regime in which positive thrust is created in the presence of a translational velocity is called the <u>propeller regime</u> (Figure 178). This is the basic and most important operating regime and is used during taxi, takeoff, climb, horizontal flight, and to some degree during glide and landing. In Figure 176 this flight regime corresponds to the segment ab, excluding points a and b. The thrust and power coefficients decrease as the advance ratio λ increases. At the same time the propeller efficiency initially increases, reaches a maximum at point b, and then decreases rapidly. The point b characterizes the optimal propeller operating regime for a given value of the blade incidence angle. Thus, positive values of the coefficients $\bar{\alpha}$, $\bar{\beta}$ and η correspond to the propeller regime of propeller operation.

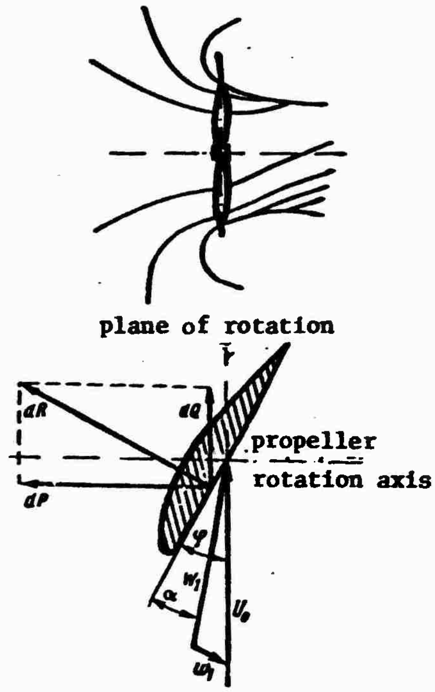


Figure 177. Flow past propeller, and blade element velocities and aerodynamic forces with propeller operating statically.

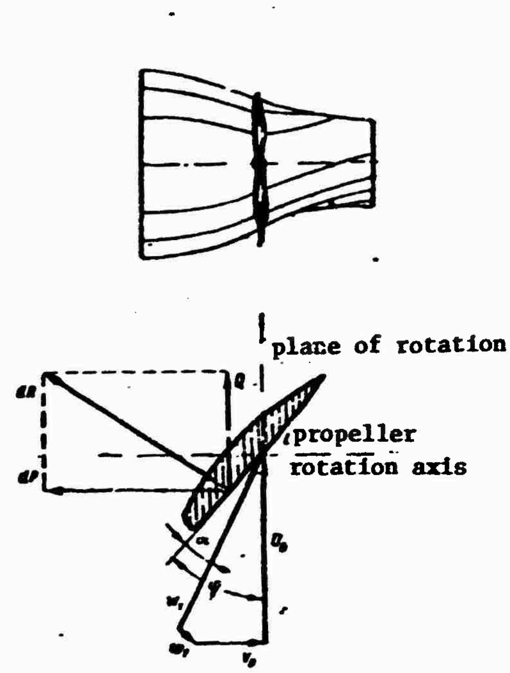


Figure 178. Flow past propeller and blade element velocities and aerodynamic forces with propeller operating in propeller regime.

The operating regime in which the propeller does not create either positive or negative thrust (drag) is termed the zero thrust regime. In this regime the propeller essentially screws itself freely into the air, without forcing the air back and not creating any thrust (Figure 179). In Figure 176 the zero thrust regime corresponds to point c. Here the propeller thrust coefficient $\bar{\alpha}$ and the efficiency are zero. The power coefficient $\bar{\beta}$ has some positive value. This means that engine power is required to overcome the propeller rotational resistance torque in this regime.

The zero thrust regime can occur during airplane glide. In this case the blade angle of attack is usually somewhat less than zero.

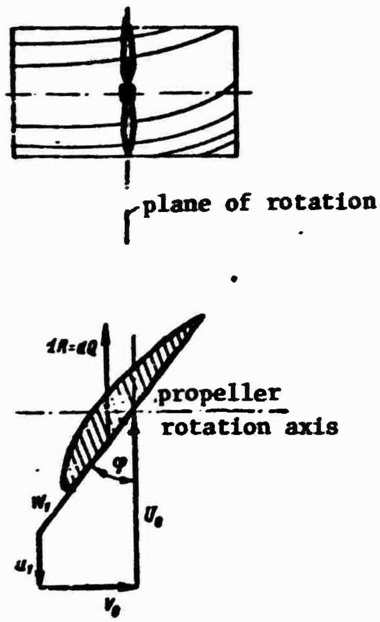


Figure 179. Flow past propeller, and blade element velocities and aerodynamic forces with propeller operating in zero-thrust regime.

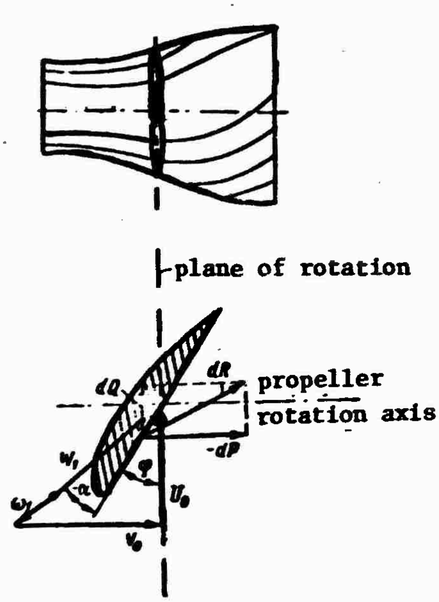


Figure 180. Flow past propeller, and blade element velocities and aerodynamic forces with propeller operating in the brake regime.

The propeller operating regime in which negative thrust (drag) is created with positive power on the engine shaft is termed the <u>propeller brake regime</u> (Figure 180). In this regime the inflow angle β is larger than the incidence angle ϕ, i.e., the blade angle of attack α is negative. In this case the airstream exerts pressure on the back of the blade and therefore creates negative thrust. In Figure 176 this propeller operating regime corresponds to the segment between points c and d, where the coefficients $\bar{\alpha}$ and η have negative values, while the values of the coefficient $\bar{\beta}$ vary from some positive value to zero. As in the preceding case, engine power is required to overcome the propeller rotational resistance torque.

Negative propeller thrust is used to shorten the landing ground roll distance. For this purpose the blades are specifically set to a minimal incidence angle ϕ_{min} such that the angle of attack α is negative during airplane landing rollout.

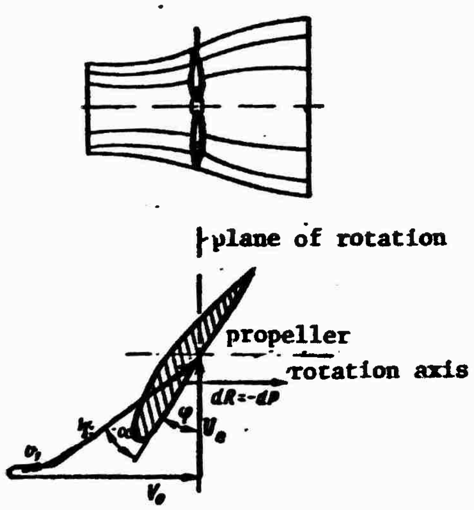


Figure 181. Flow past propeller, and blade element velocities and aerodynamic forces with propeller operating in the autorotation regime.

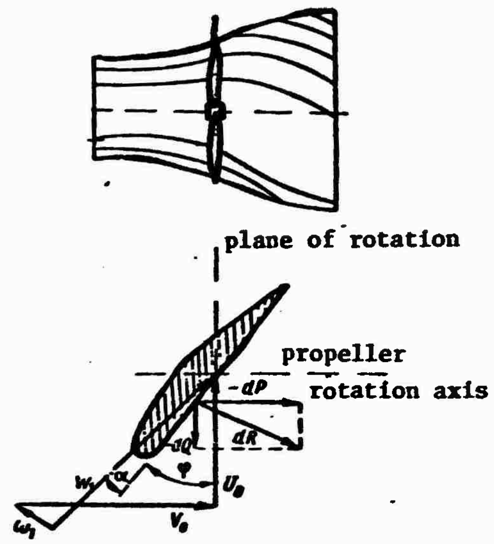


Figure 182. Flow past propeller, and blade element velocities and aerodynamic forces with propeller operating in the windmill regime.

The operating regime in which the engine shaft power is zero and the propeller rotates at the expense of the relative airflow energy (under the influence of the aerodynamic forces applied to the blades) is termed the <u>autorotation regime</u> (Figure 181). In this case the engine develops only the power required to overcome the internal friction forces and torques which are generated as the propeller rotates. In Figure 176 this regime corresponds to point d. The propeller thrust is negative, just as it is in the brake regime.

The operating regime in which the engine shaft power is negative and the propeller rotates at the expense of the relative airstream energy is termed the <u>windmilling regime</u> (Figure 182). In this regime the propeller not only does not require engine power, but the propeller itself rotates the engine shaft at the expense of the relative airstream energy. In Figure 176 this regime corresponds to the segment to the right of point d. The windmilling regime is used for restarting the engine in flight. In this case the engine shaft is accelerated to the rotational speed required for starting without requiring any special starting equipment.

Airplane deceleration during the landing rollout also begins in the windmilling regime and passes sequentially through the stages of autorotation and braking to the zero thrust regime.

Classification

We have shown previously that the magnitude of the blade angle of attack for a constant setting ϕ depends on the flight speed. This phenomenon applies to the fixed pitch propellers. Their primary drawback is that during takeoff (at low flight speed) they may be "heavy" and will not permit the engine to develop takeoff power.

Conversely, in horizontal flight at high translational speed the propeller is "light," and in this case the rotational speed increases to unacceptably high values, at which reliable engine operation is not assured.

In the past, when the airplane flight speed range was small, fixed pitch propellers (FPP) were used. As airplanes were improved and the speed range increased, propellers with pitch variable in flight (VPP) began to be used. The first propellers of this type had a comparatively small range of variation of the incidence angle, usually not exceeding 10°. As a rule these were two-pitch propellers. In this case the takeoff and climb were made at a low setting (low pitch) which made it possible to obtain takeoff engine rpm at zero speed on the ground. Upon transition to horizontal flight, the blades were changed to high pitch by opening the valve of a special hydraulic system or by means of a wheel which provided mechanical control of the propeller.

With further increase of the airplane flight speed range and, therefore, with increase of the incidence angle variation range, use began to be made of propellers with automatic systems for regulating rpm by varying the incidence angle as a function of the flight regime.

Propellers with such rpm control systems are termed automatic air propellers. The construction of these propellers involves very complex assemblies, whose successful operation and maintenance are possible only if a thorough study is made of their operation and the operating instructions.

FORCES AND MOMENTS ACTING ON BLADES

Blade Centrifugal Forces

As the propeller rotates, centrifugal forces, directed from the axis of rotation and perpendicular to this axis, act on the blade elements. The magnitude of the centrifugal force of any blade element is

$$dP_c = \omega^2 r dm, \tag{108}$$

where dm is the element mass.

The blade element mass may be found from the formula

$$dm = \frac{\gamma}{g} s dr,$$

where γ is the specific weight of the material;

s is the blade section area;

g is the gravitational acceleration.

Substituting the value of dm into (108), we obtain

$$dP_c = \frac{\gamma}{g} \omega^2 s r dr.$$

Integrating this equation over the entire blade length in the limits from r_0 to R, we find the magnitude of the blade centrifugal force

$$P_c = \frac{\gamma}{g} \int_{r_0}^{R} \omega^2 s r dr.$$

For modern propellers the magnitudes of the centrifugal forces reach very large values. These forces are taken by the propeller hub.

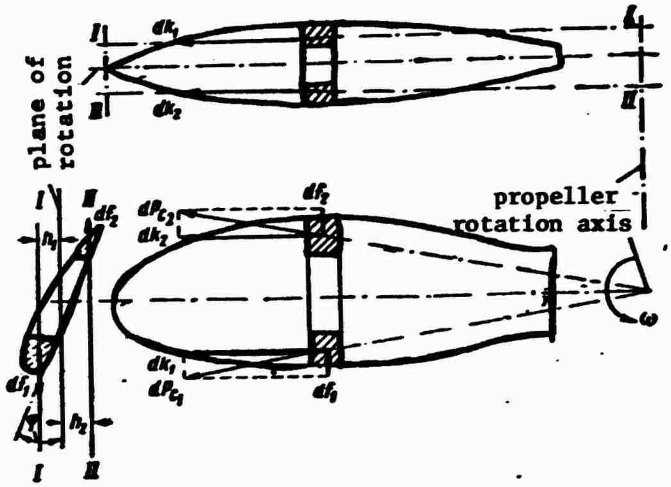


Figure 183. Action of blade element centrifugal forces.

Let us take two blade elements which lie between two parallel sections, one at the leading edge, the other at the trailing edge (Figure 183). The centrifugal force vectors dP_{c_1} and dP_{c_2} of these elements are directed away from the axis of rotation and perpendicular to this axis. They can be resolved in planes parallel to the propeller rotation plane along directions parallel (dk_1 and dk_2) and perpendicular (df_1 and df_2) to the blade axis. These forces are also shown on the blade cross section.

If we resolve the centrifugal force vectors for the other elements located between the leading and trailing edges at the same blade section, we obtain the pattern of the centrifugal force transverse components (Figure 184). The vectors comprising this pattern are located in planes perpendicular to the propeller rotation axis.

We see from Figure 184 that the centrifugal force transverse components change their direction upon transition through the blade axis, which is usually also the line of centers of gravity of the sections. We replace the forces in each direction by the equivalent forces dF_1 and dF_2; then the action of the entire force pattern will be equivalent to a couple. The magnitude of the couple will change

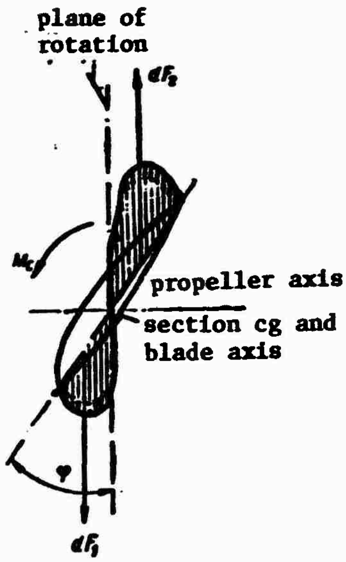


Figure 184. Distribution of centrifugal force transverse components across blade section.

its value along the blade. The total couple for the entire blade is the quite large moment M_c from the centrifugal force transverse components, which twists the blade in the direction of decreasing incidence angle.

In variable pitch propellers blade rotation to the required incidence angle is accomplished about axes which coincide with the axes of the butt (cylindrical) parts of the blades. The magnitude of the moment M_c, tending to decrease the incidence angle, depends on propeller rpm, material, geometric dimensions, and blade angles of incidence and twist, i.e.,

$$M_c = f(\omega, \varphi, \gamma, b, c, R, q, \bar{\varphi}),$$

where c is the blade thickness;

q is a coefficient which depends on the blade profile type.

The magnitude of the blade centrifugal force transverse moment is considered both in designing and operating the systems for automatic engine rpm regulation.

Aerodynamic Forces

The aerodynamic forces arise as a result of the action of the airstream on the blades, and are distributed over the entire blade surface. This blade loading scheme can be considered as a beam which is fixed at one end and subjected to the action of a distributed loading which creates bending and twisting moments.

The resultant dR of the blade element aerodynamic forces is applied at the center of pressure, which is usually located ahead of the blade rotation axis (see Figure 175). The force dR on the arm l or forces dP and dQ on the arms a and b create the twisting moment $dM_a = dRl = dPa + dQb$ about the blade rotation axis, tending to turn the blade in the direction of increasing incidence angle. The magnitude of the overall aerodynamic force moment of the entire blade for a given propeller depends on the blade angles of attack and the resultant relative airstream velocities. In view of the comparatively small values of the arm l, the magnitude of the aerodynamic moment is not large.

In the case of negative blade angles of attack the direction of the resultant force dR changes so that the aerodynamic twisting moments in this case tend to turn the blades in the direction of decreasing incidence angle.

Counterweight Centrifugal Forces

Since the aerodynamic twisting moment is not large, it cannot be used as an independent source of energy for rotating the blades in the increasing incidence angle direction.

In this connection some propellers are equipped with additional counterweights, which are mounted on brackets on the blade shank (Figure 185).

As the propeller rotates, the counterweight centrifugal forces P_{cw} arise, directed away from the axis of rotation. The counterweights are located relative to the blades so that the components P_{cwf} create on the arm h the blade twisting moment $M_{cw} = P_{cwf}h$, which tends to turn the blade in the direction of increasing incidence angle. The counterweight can be located either above or below the blade (latter arrangement is shown dashed).

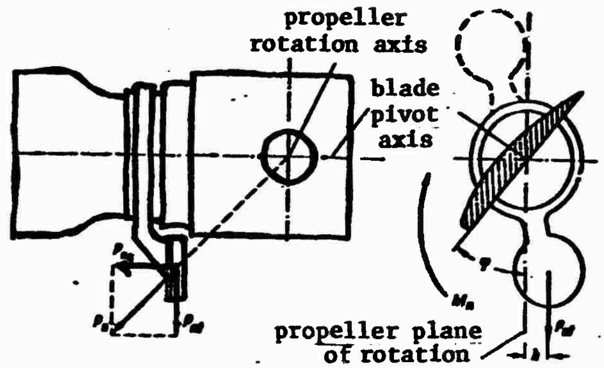


Figure 185. Action of counterweight centrifugal force moment.

The magnitude M_{cw} of the counterweight twisting moment depends on its mass, distance from the axis of rotation, arm h, and propeller rpm. All these parameters are selected so that the combined action of the two twisting moments from the counterweight centrifugal forces and the aerodynamic forces will provide blade rotation in the direction of increasing incidence angle with the required rate of rotation.

The counterweight force component P_{cwk}, directed along the blade, gives rise to an additional bending moment, which is taken by the counterweight bracket.

Hydraulic Propellers

Hydraulic propellers are those in which the incidence angle is changed by liquid pressure on a piston (Figure 186). Depending on the position of the distribution valve 5, oil can be supplied by the gear pump 6 to the cylinder 1 on either side of the piston 13. The piston is connected with the propeller blade 10 by the rod 12 and the arm 11, located eccentrically on the end of the blade sleeve. Translational motion of the piston rotates the blade sleeve. The blade is restrained on ball bearings in the hub. The aviation oil which is used for engine lubrication is also used as the working fluid in such systems.

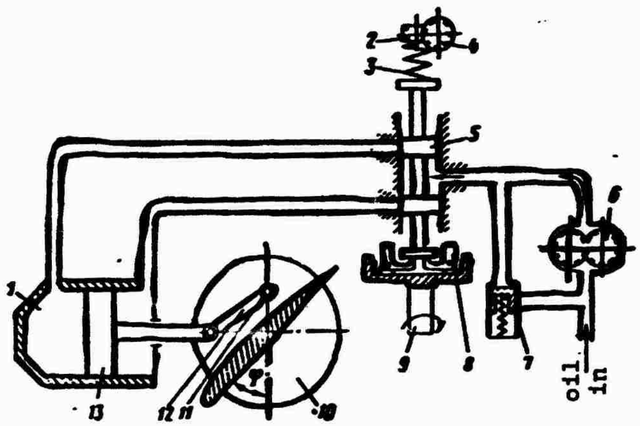


Figure 186. On-speed operation of propeller and governor: 1 - cylinder; 2 - rack; 3 - spring; 4 - pinion; 5 - distributor valve; 6 - pump; 7 - relief valve; 8 - weights; 9 - shaft; 10 - blade; 11 - drive arm; 12 - connecting rod; 13 - piston.

The functional schematic shown in Figure 186 can be realized in many ways. We shall examine the most characteristic cases of its application for several hydraulic propellers.

Since the blades of an operating propeller are acted upon by the centrifugal force transverse component moments, tending to turn the blade in the direction of decreasing incidence angle, and by the aerodynamic moments, tending to turn the blade in the opposite direction, it is obvious that in order to rotate the blade in both directions we must have a system which can create additional controlled moments which are transmitted through a rotating mechanism to the propeller blades. In systems with hydraulic mechanisms these moments are created by oil pressure force. With regard to operating scheme, the hydraulic propellers are subdivided into direct, reverse, and two-way action propellers.

The <u>direct action propeller</u> is one in which rotation of the blades in the direction of decreasing incidence angle (pitch) is accomplished by the sum of the moments of the oil pressure force and the blade centrifugal force transverse components, while rotation in the direction of increasing incidence angle is accomplished by the

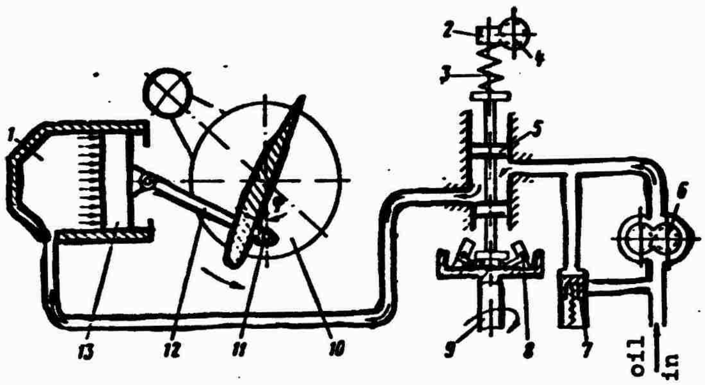


Figure 187. Underspeed operation of direct-action propeller and governor (legend same as Figure 186).

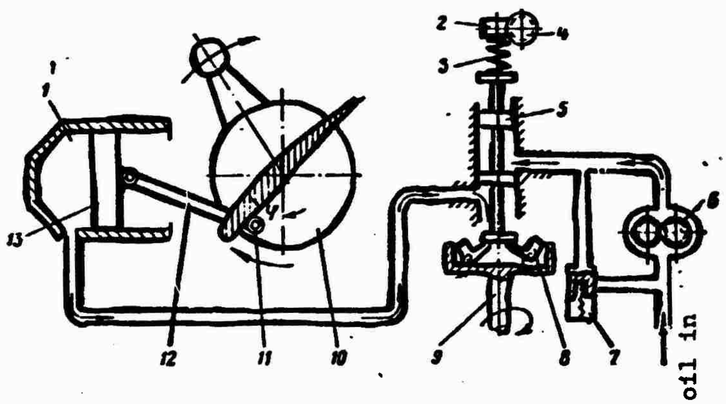


Figure 188. Overspeed operation of direct-action propeller and governor (legend same as Figure 186).

sum of the moments of the counterweight centrifugal force transverse components and the aerodynamic forces (Figures 187 and 188). Rotation of the blades in the direction of decreasing incidence angle takes place when the following inequality holds

$$M_o + M_c > M_{cw} + M_a + M_{fr},$$

where M_o is the moment of the oil pressure force;

M_{fr} is the moment of the friction forces in the blade rotation mechanism.

Increase of the incidence angle takes place when the following inequality holds

$$M_{cw} + M_a > M_o' + M_c + M_{fr},$$

where M_o' is the moment of the resistance force of the oil which is displaced from the cylinder.

The moment of the oil pressure force can change in magnitude as a function of the position of the spool valve. This provides the desired relationship of the moments for blade rotation in either direction. Clearly, when the algebraic sum of all the moments acting on the blades equals zero ϕ = const, i.e., a constant incidence angle is maintained.

The reverse-action propeller is one in which rotation of the blades in the direction of decreasing incidence angle is accomplished by the moment of the centrifugal force transverse components, while the rotation in the direction of increasing incidence angle is accomplished by the sum of the moments of the oil pressure force and the aerodynamic forces (Figures 189 and 190). Pitch reduction of the reverse-action propeller takes place when

$$M_c > M_a + M_{fr} + M_o',$$

and pitch increase takes place when

$$M_o + M_a > M_c + M_{fr}.$$

The two-way-action propeller is one in which rotation of the blades in the direction of decreasing incidence angle is accomplished by the sum of the moments of the oil pressure force and the blade centrifugal force transverse components, while rotation in the direction of increasing incidence angle is accomplished by the sum of the moments of the oil pressure force and the aerodynamic forces (Figures 191 and 192). By analogy with the preceding propellers, we write the

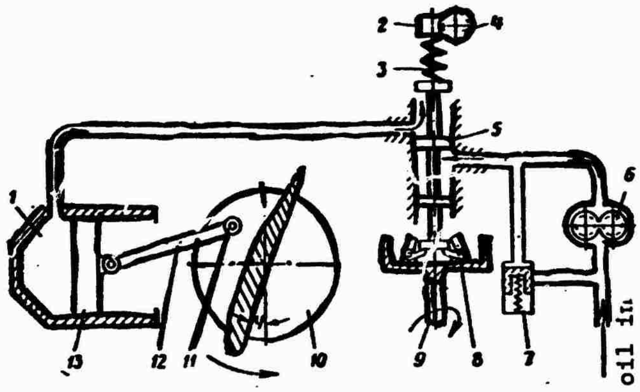


Figure 189. Underspeed operation of reverse-action propeller and governor (legend same as Figure 186).

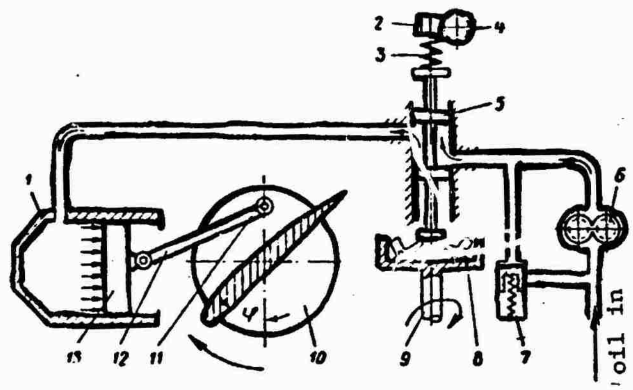


Figure 190. Overspeed operation of reverse-action propeller and governor (legend same as Figure 186).

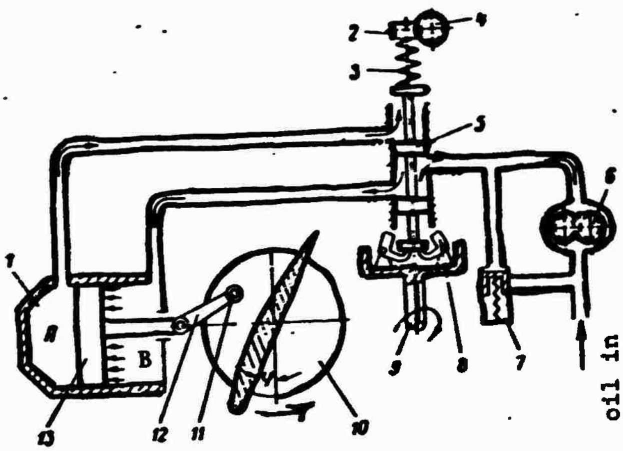


Figure 191. Underspeed operation of two-way-action propeller and governor (legend same as Figure 186).

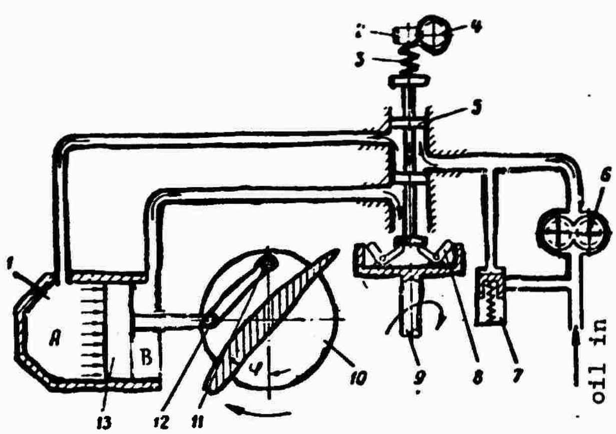


Figure 192. Overspeed operation of two-way-action propeller and governor (legend same as Figure 186).

condition for pitch decrease of the two-way-action propeller

$$M_{o_B} + M_c > M_a + M_{fr} + M_{o_A}$$

and the pitch increase condition

$$M_{o_A} + M_a > M_c + M_{fr} + M'_{o_B}$$

where M_{o_B} is the oil pressure in chamber B;

M_{o_A} is the oil pressure in chamber A.

In the general case the oil pressure moments M_{o_A} required to increase and M_{o_B} to decrease propeller pitch are not the same. This is explained by the fact that the moment of the blade centrifugal force transverse components is considerably larger than the moment of the aerodynamic forces. Therefore the moment required from the oil pressure force for increasing pitch is greater than for decreasing pitch. In this connection the oil supply into the two chambers of the propeller cylinder can be provided from different sources (pumps) and therefore at different pressures.

In accordance with the definition of the two-way-action propeller, the propellers of the first two schemes are termed single-action propellers, since in these propellers the blades are rotated in one direction only by the moment from the oil pressure force.

In case of oil system failure which results in reduction of the oil pressure, the reverse-action and two-way-action propellers have a tendency to decrease pitch and the engine rotor tends to increase speed. This phenomenon is very undesirable, since it can lead to engine failure. The direct-action propellers are free of this drawback. When the oil pressure drops off, the propeller blades are rotated by the moments of the centrifugal force transverse components in the direction of increasing pitch angle, i.e., the blade loads up. However, the presence of the counterweights increases propeller weight.

The hydraulic systems provide high blade rotation rates, since the moments from the oil pressure forces can reach large values, and these systems are comparatively simple in construction and reliable in operation. This explains the wide use of hydraulic propellers.

Electromechanical Propellers

The electromechanical variable pitch propeller is one in which rotation of the blades 1 (Figure 193) is accomplished by an electric motor 2. The electric motor is reversible, since the blades must be rotated to both increase and decrease the pitch. Transmission of the torque from the electric motor to the propeller blades is accomplished through a reduction gearing with very large reduction ratio. The electric motor is usually mounted on the front part of the propeller. The current is supplied to the motor from the ship's system through slip rings attached to the aft side of the hub and brushes.

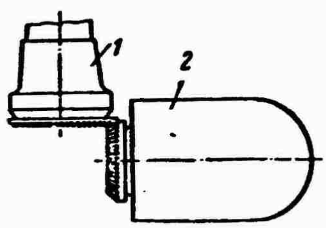


Figure 193. Use of electric motor to change blade pitch angle.

The advantages of propellers of this type include unlimited capability for selection of the blade rotation range, since the blades can rotate about their axes through 360°. Moreover, it is possible to de-energize the electric motor power system and thereby lock the blades in position when necessary, i.e., the automatic VPP can be made into a FPP. The necessity for this may arise in emergency situations in flight and when performing adjustments on the ground (rpm adjustment, mixture adjustments, ignition system checks, and so on).

However, in spite of these advantages electromechanical propellers are not used on Soviet aircraft. This is explained first of all by the fact that they are less reliable in operation, more complex and expensive to fabricate in comparison with the hydraulic propellers. One of the real drawbacks of propellers of this type is the low blade rotational speed, which leads to large engine rpm overshoots and sometimes to propeller overspeeding in the transient regimes. This drawback is associated with the use of electric motors of limited power. Increase of the blade rotation speed requires increase of the electric motor power, and this increases its size and weight markedly.

Aeromechanical Propellers

Aeromechanical propellers, in which blade rotation is accomplished automatically without the use of outside energy sources or governor, are used on airplanes with low-power engines. Therefore these propellers are self-contained and automatic. Automatic blade rotation is achieved by varying in flight the magnitude of the torques acting on the propeller blades.

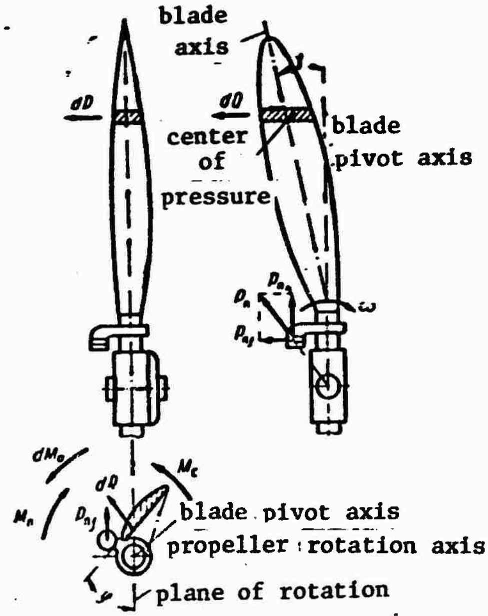


Figure 194. Aeromechanical propeller.

We have shown previously that the aerodynamic forces create torques about the blade pivoting axis. In conventional propellers the magnitude of these torques is small and the direction of their action is determined by the magnitude of the angles of attack. If the blades are given a special shape or they are deflected through the angle (Figure 194) relative to the blade pivot axis, then as a result of the center of pressure location change the moments of the aerodynamic forces will tend to rotate the blades in the direction of decreasing incidence angle.

Like the direct-action propellers, the blades of the aeromech-nical propellers are equipped with counterweights, which create torq.. - directed toward increase of the blade pitch angle. The blade centrifugal force transverse component moments M_c tend to rotate the blades in the direction of decreasing blade pitch angle. The torques M_{cw} created by the counterweights are larger than the torques created by the blade centrifugal force transverse components. Under steady state conditions the relation $M_{cw} = M_c + M_a$ must be satisfied.

However, the moment values discussed above vary as a function of the flight regime; therefore selection of the proper relationship between the torques acting on the propeller blades over a wide range of pitch angle variation is a very important and complex task. This relationship between the torques must provide for blade pitch increase with increase of the flight speed and, conversely, blade pitch decrease with decrease of the flight speed. At a fixed engine operating condition, the rotational speed should remain constant.

Accordingly, during static engine operation, when the propeller thrust is maximal and therefore the torque from the aerodynamic forces is maximal, the propeller blades are set on the low-angle stop. This makes it possible to obtain takeoff engine rpm and optimal airplane takeoff conditions. Then, as the flight speed increases during takeoff and climb, the propeller thrust decreases and the moments M_a decrease, while the torques from the centrifugal forces of the counterweights and blades, which are independent of the flight speed, retain their previous values (if $n = const$). As a result of this the relationship between the torques changes and the blades will gradually rotate in the direction of increasing pitch angle, thus preventing propeller overspeeding. In horizontal flight the speed increases still more and the propeller thrust decreases to a greater degree, which leads to further propeller pitch increase. This picture is obviously reversed when the flight speed decreases. Thus, the blades of the aeromechanical propeller change their pitch angles automatically as a function of the flight speed. In this procedure the propeller rotational speed changes, but within comparatively narrow limits. Variation of the engine operating regime is achieved by changing the fuel flowrate.

The advantages of this propeller type include: manufacturing and operational simplicity, low weight and small size of the propeller hub; while the disadvantages include reduction of the set rpm as the airplane climbs, which causes reduction of the engine power. It follows from (107) that propeller thrust decreases with climbs in altitude because of the reduced air density. This causes the blade pitch angle to increase and engine rpm and power to decrease. Serious difficulties also arise in providing the required relationship between the torques over a wide range of variation of the flight conditions. These drawbacks of the aeromechanical propellers narrow their field of application significantly.

Mechanical Propellers

Mechanical propellers are those in which blade rotation is accomplished with the aid of a special mechanism which is operated by pilot muscular effort or by crankshaft rotational energy. Prior to takeoff the pilot sets the blades at the minimal angle, thereby ensuring optimal takeoff conditions. As the flight speed increases, the pilot increases the propeller pitch angle and thus maintains the specified engine rpm. Thus, with change of the flight conditions the pilot must himself change the blade setting angle every time. This is the primary drawback of the mechanical propellers and is the reason for their very limited application.

Combined Operation of Propeller and Governor

Hydraulic propellers. Only automatic propellers are used on the modern turboprop powered airplanes, and these require in addition to the elements discussed above the installation of governors with a centrifugal type sensor (see Figure 186). The purpose of the governor in combination with the variable pitch propeller is to maintain automatically the specified constant engine rpm. This speed is set by adjusting the tension on the governor spring 3 and is monitored on the tachometer.

Let us assume that some rpm has already been set in the governor. This rpm is automatically maintained by the propeller-governor system as follows. Two forces act at all times during engine operation on the governor slide valve 5: the elastic force of the spring 3, tending to force the valve downward, and the centrifugal forces of the weights 8, which tend to drive the weights outward so that their shoulders lift the valve upward. If the engine is operating in a steady-state regime, in which the rpm is maintained constant, the slide valve 5 is in the neutral position (the oil flow passages are blocked by the valve lands), and equilibrium is established between the spring elastic force and the weight centrifugal forces. The engine rpm corresponding to this position is termed the equilibrium or preset speed. It is obvious that the greater the spring compression, the larger the weight centrifugal forces and the higher the engine rpm required to keep the valve in the neutral position, and vice versa.

Now let us assume that the engine rpm has decreased for some reason, i.e., it is less than the desired speed. This can be either because of decrease of the fuel flowrate or because of decrease of the flight speed, for example when changing from level flight to climb. In the first case the engine power decreases and for a given blade pitch angle becomes insufficient to turn the propeller at the previous rpm. In the second case the blade angle of attack increases because of the flight speed reduction and the propeller becomes "heavier"; therefore the engine cannot turn the propeller at the same rpm. In both cases the governor must lighten the propeller, i.e., reduce the pitch angle to a value at which the specified rpm will be restored. This is achieved as follows. As the rotor speed decreases, the centrifugal forces of the weights decrease, and this causes the slide valve to move downward from the equilibrium (neutral) position.

In the direct-action propellers the pump supplies oil in this case into the cylinder group, where the piston moves to the right as a result of the pressure increase (see Figure 187). Since the piston is connected with the blades by means of rods and arms located eccentrically on the blade shanks, as the piston translates to the right the blades rotate in the direction of lower pitch angle. As this happens the rpm increases. The propeller unloading process terminates when the specified rpm is reached, i.e., when the slide valve reaches the neutral position and blocks the oil passage connecting the propeller cylinder chamber with the governor pump.

In the reverse-action propellers (see Figure 189) the cylinder oil chamber is connected with the passage which returns the oil into the engine case when the rpm decreases and the slide valve moves downward from the neutral position. In this case the blades turn in the direction of decreasing pitch angle under the influence of their own centrifugal force transverse components, with the piston displacing the oil from the cylinder group. The propeller unloading process terminates as the specified r.. is reached, when the governor slide valve moves upward and blocks the oil return passage.

In the two-way-action propellers (see Figure 191) the slide valve moves down as the rpm decreases and opens up two passages. In this case the oil supplied by the pump enters chamber B of the cylinder group and moves the piston to the left. The oil is forced from chamber A by the piston through the open upper passage and returns to the case. As in the preceding cases, the propeller unloading process terminates as the preset rpm is reached, when the slide valve blocks both oil passages.

Now we shall examine in the same sequence the combined operation of the propeller and governor for all three systems in the case when the rpm increases above the selected speed.

In flight the rpm increases when the power lever is moved forward (when the fuel flowrate is increased) or when the flight speed increases, for example when transitioning from the climb to level flight. In the first case the engine power increases and the propeller becomes "light" if the previous blade pitch angle is retained. There is an excess of the power developed by the engine over the power required by the propeller. In the second case (increase of the flight speed at a constant position of the power lever), the blade angle of attack decreases and the propeller becomes "lighter." In both cases the engine rpm tends to increase and the governor must load up the propeller and thereby retain the preset rotational speed.

In the direct-action propellers (see Figure 188) this is accomplished as follows. As a result of the rpm increase, the centrifugal forces of the weights increase and become larger than the spring elastic force. Therefore the slide valve moves up and provides free return of the oil from the cylinder group to the engine case. In this case, under the action of the counterweight centrifugal force transverse component moment and the aerodynamic moments the blades rotate in the direction to increase the pitch angle to restore the preset rpm. When this occurs the slide valve moves down and blocks the oil return line from the cylinder group.

In the reverse-action propellers (see Figure 190), as the rpm increases the governor slide valve moves up and opens the passage through which oil from the pump enters the cylinder group. In this case the piston moves to the right and rotates the blade in the direction of increasing pitch angle. The propeller loading process terminates when the preset rpm is reached and the slide valve moves down to block the oil passage.

Upon increase of the rotational speed, the two-way-action propellers are loaded up as follows. As the slide valve moves up, it opens two oil passages (see Figure 192). Oil from the pump flows through the upper passage into cylinder chamber A and displaces the piston, which in turn, being connected with the blades, rotates the latter in the direction of increasing pitch angle. The oil returns to the engine case through the lower passage from chamber B. As in the preceding cases, the propeller loading process terminates when the slide valve blocks the oil passages as it moves downward.

The variable pitch electromechanical propeller, operating together with an electrohydraulic or electric governor, automatically maintains the preset engine rpm. When there is a change of the preset rotational speed, the governor closes with the aid of a relay and electric circuit from a storage battery to the electric motor. The latter is mounted on the propeller and is activated when the electric circuit is closed and thus turns the blades in the desired direction. In case of automatic control system failure manual control may be used.

Propeller Negative Thrust and Means for its Prevention in Flight

Conditions for Occurrence of Negative Thrust

Propeller negative thrust occurs in flight whenever the blade angle of attack takes negative values. On airplanes having piston engines with variable pitch propellers, the magnitude of the negative thrust is comparatively small. This is explained by the fact that the minimal blade pitch angles ϕ_{min} for these airplanes are selected in the range 19-22°. The range of pitch angle variation in flight is determined by the flight speed variation range and usually does not exceed 30-35°. If for any reason the rotational speed governing system fails in flight and the blades go to the minimal pitch angle, negative propeller thrust can occur but its magnitude for these values of ϕ_{min} does not reach large magnitudes, and therefore does not lead to significant complications in control of the airplane.

Prevention of the development of negative thrust in flight for airplanes with turboprop engines is one of the most critical and important problems. In case of engine failure or malfunctions in the rotational speed governing system, the propeller negative thrust may reach very large values, sometimes exceeding in magnitude the maximal positive thrust. Sudden onset of negative thrust in flight if the flight crew is not adequately prepared can lead to serious consequences, since in this case the airplane yaws sharply and banks in the direction of the failed powerplant.

The possibility of large negative propeller thrust developing on airplanes with TPE is explained primarily by the fact that the geometric and aerodynamic characteristics of the propellers on airplanes with TPE differ considerably in comparison with the propellers on airplanes with PE. The propellers for the TPE have considerably larger dimensions (blade width, diameter, and number) than those for the PE. At the same blade pitch angle the negative thrust of the propeller on the airplane with a TPE is greater than on an airplane with PE.

Another reason for the appearance of large negative thrust on airplanes with TPE is the existence of a large range of blade rotation angles and small values of the minimal pitch angles. As is well known, in order to reduce the required starting power the propeller minimal blade angle ϕ_{min} of the single-spool TPE is usually in the range 0-8°, which provides minimal resistance to propeller rotation during engine starting and a small value of the thrust when operating on the ground at idle power. Transition of the blades to these angles during the rollout after landing also makes it possible to obtain the required negative thrust to shorten the landing roll.

However, in the case of engine failure in flight the governor, attempting to maintain the preset rotational speed, drives the blades to small pitch angles (negative angles of attack), at which large negative thrust develops and makes it difficult to continue flight. Drift of the blades to low pitch angles and the occurrence of negative thrust can also happen if the rotational speed governing system fails, when in the absence of oil pressure in the propeller cylinder group the blades, not experiencing any counteraction, are driven to ϕ_{min}

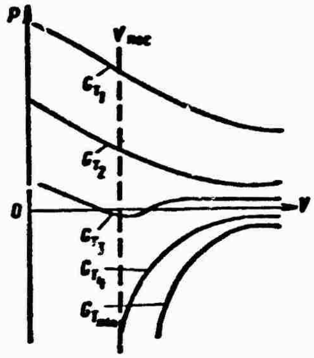


Figure 195. TPE thrust variation as function of fuel flowrate and flight speed.

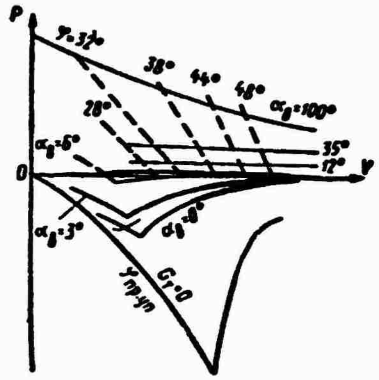


Figure 196. Variation of thrust and blade pitch angle as functions of flight speed for various positions of engine fuel control lever.

by their own centrifugal force transverse component moments.

In Figure 195 curve G_{f_1} corresponds to rated power, G_{f_2} — cruise power, G_{f_3} — flight idle power. G_{f_4} is for an intermediate regime between flight and ground idle, and $G_{f_{min}}$ is for ground idle power. These curves apply to flights at low altitude (sea level).

We see from these curves that reduction of the fuel flowrate to G_{f_4} or below causes the appearance of negative thrust, which is the greater the lower the flight speed. However, this is valid only down to a definite velocity, at which the maximal negative thrust is reached and the blades come up against the stop (intermediate or ϕ_{min}). With further decrease of the flight speed the negative thrust decreases (Figure 196). The maximal value of the negative thrust occurs with engine failure. This pattern of thrust variation as a function of flight speed is characteristic only for propellers with automatic pitch variation.

For fixed-pitch propellers the magnitude of the negative thrust which develops when an engine fails varies in proportion to the change of the flight speed. With increase of the flight speed, the autorotational speed and the negative thrust of the propeller increase. These parameters decrease as the flight speed is decreased.

To explain the relations governing negative thrust variation in flight for the variable-pitch propellers, we turn to analysis of the governing system operation. We assume that the governor has a fixed setting corresponding to the program n = const.

We noted previously that the magnitude of the pitch angle is determined by the fuel flowrate (by the position of the lever controlling the fuel supply to the engine) and by the flight speed. The higher the fuel flowrate and the flight speed, the larger is the pitch angle, and vice versa. In the case in which engine power decreases, the governor moves the propeller blades to lower pitch angles in order to maintain n = const. In this case the propeller transitions from the propeller regime into the windmilling regime and the propeller-turbocompressor system will autorotate. If the flight speed is high, the rotational speed in autorotation in the case of engine shutdown may not decrease and may remain under control of the governor. In this case the governor will not permit the blades to move to ϕ_{min} but keeps them at some intermediate angle. The magnitude of this angle is smaller than that existing prior to engine shutdown and depends on the flight speed. The higher the flight speed, the larger the pitch angle during autorotation of the propeller-turbocompressor system, and vice versa. It is obvious that when the pitch angles at high flight speeds exceed ϕ_{min} considerably the propeller negative thrust is significantly less than its maximal value, which corresponds to ϕ_{min}.

At low flight speeds, when the rotational speed in autorotation does not reach the rotational speed to which the governor is set, the propeller escapes from governor control (Figure 197) and very large negative thrust develops, since the blades go to ϕ_{min}.

We see from the figure that at the first instant after engine shutdown the rotational speed decreases slowly. The propeller blades

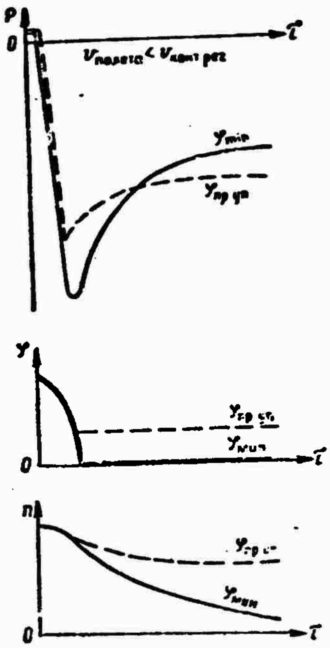


Figure 197. Variation of TPE negative thrust, blade pitch angle, and rpm during enginr failure (or shutdown) in flight.

go to the angle ϕ_{is}, but as a rule in this case there is a lag in the operation of the governing system, which leads to reduction of the engine rpm. Then, after the blades reach the angle ϕ_{is}, the rotational speed begins to reduce more rapidly. The propeller becomes a fixed-pitch propeller and autorotates under the influence of the relative airstream. The rotational speed in autorotation and the negative thrust are determined by the magnitude of the flight speed, increasing with increase of the latter and vice versa. The negative thrust is maximal at the first instant of stable autorotation, when the blades reach the stop ϕ_{is} while still having high rotational speed. This phenomenon is usually termed "negative thrust overshoot." Thereafter the value of the negative thrust decreases as the propeller transitions to stable autorotation and the rotational speed decreases.

Thus, in the case of powerplant failure there is a region of high speeds in which the propeller autorotates under the control of the governor, and a region of low speeds in which the propeller escapes form governor control and the blades go to the intermediate stop. In the latter case the VPP becomes a FPP.

The flight altitude affects the magnitude of the negative thrust. The lower the flight altitude, other conditions being the same, the higher the negative thrust is, and, conversely, with increase of the flight altitude the negative thrust decreases (Figure 198). These phenomena are explained by variation of the air density. The magnitude

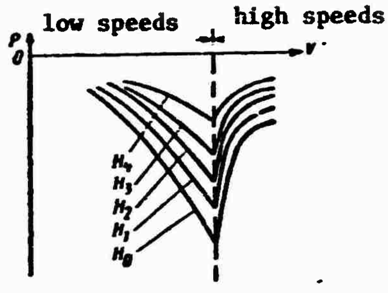


Figure 198. Negative thrust as function of flight speed and altitude ($H_4 > H_3 > H_2 > H_1 > H_0$).

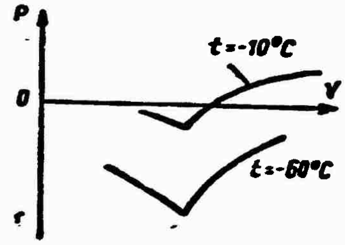


Figure 199. Effect of ambient air temperature on magnitude of negative thrust ($\alpha_{th} = 6°$).

of the negative thrust also depends on the ambient air temperature (Figure 199). With decrease of the ambient air temperature there is an increase of the compressor power required, since the mass of air which the compressor must compress increases. But at a fixed fuel flowrate the power of the turbine remains constant and therefore is not sufficient to rotate the compressor and propeller. In this case the governor unloads the propeller, moving the blades to lower pitch angles. In certain flight regimes, particularly in a descent when the engines are throttled, the blades may go to a pitch angle at which the angle of attack will be negative and therefore negative thrust develops. The negative thrust will be greater the lower the ambient air temperature.

The occurrence of negative thrust in the prelanding glide is very undesirable, since it may complicate airplane landing significantly and may even lead to an accident. To avoid this situation it is necessary that the power lever detent be varied as a function of the ambient air temperature, shifting the detent in the direction of increasing fuel flowrate as the air temperature decreases and vice versa. In this connection, prior to landing the pilot must request the air temperature at the landing airfield and set the detent to the position corresponding to this temperature on the scale on the engine control pedestal.

Negative thrust can also occur if the propeller blades ice up, since in this case the power required to rotate the propeller increases. Since the turbine power is constant, the governor, maintaining a constant rotational speed, moves the blades to lower pitch angles and negative thrust may develop as in the preceding case.

Protective Devices

Various protective devices are provided in the engine and propeller design in order to limit the propeller negative thrust magnitude and to protect against TPE overspeeding in flight. These include pitch angle locks: hydraulic (HPL), mechanical (MPL), and centrifugal (CPL) locks; intermediate blade stop, and autofeathering based on torque, negative thrust, and limiting engine rpm. One or more of these protective devices may come into operation, depending on the airplane flight regime, engine operating conditions, and reasons for faulty operation of the propeller-governor system.

Pitch Locking

The hydraulic pitch lock consists of the valve 3 (Figure 200) installed in the oil line connecting the front chamber of the propeller cylinder with the governor. In case of proper operation of the propeller-governor system the valve 3 is always open, since oil under pressure created by the governor pump acts on the end of the valve plunger. Thus the valve 3, being in the open position, does not prevent oil entry into the cylinder during propeller pitch increase or oil exit from the cylinder during pitch decrease.

If any malfunctions develop in the governing system which involve decrease of the oil pressure, the propeller blades tend to go to the minimal angle ϕ_{min} under the influence of the moment from the centrifugal force transverse components. But in this case the pitch lock

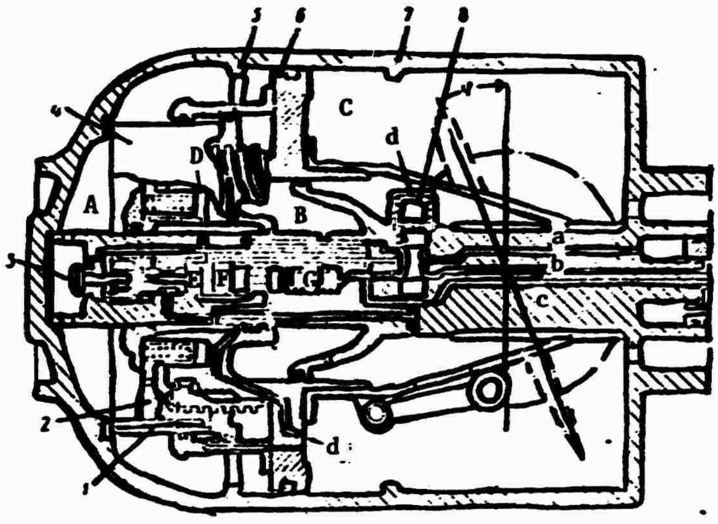


Figure 200. Schematic of blade pitch lock devices: 1 - rotating sleeve; 2 - spring sleeve; 3 - hydraulic pitch lock valve; 4 - splined sleeve; 5 - ϕ_{min} stop; 6 - piston; 7 - ϕ_{feath} stop; 8 - centrifugal pitch lock slide valve; a,b,c,d, - lock, high pitch, low pitch, and return oil passages; A,B,C,D,E,F,G - high pitch, low pitch, return, and pitch lock chambers.

valve 3 closes under the action of the spring and blocks exit of the oil from the front cylinder chamber. The oil, trapped in the cylinder, does not permit the piston 5 to move to the left and decrease the pitch angle. As flight is continued the blade pitch angle remains constant, and the propeller in essence becomes a fixed-pitch propeller. This locking of the pitch prevents the blades from going to ϕ_{min}, and therefore prevents propeller overspeeding and the development of a large negative thrust when the oil pressure in the propeller-governor system decreases.

The <u>centrifugal pitch lock</u> prevents excessive increase of engine rpm in case of governing system failure. Such failure is usually associated with decrease of the pressure in the passage supplying oil to the front cylinder chamber, or sticking of the governor slide valve in the decrease pitch position. If the engine rpm increases and exceeds the maximal permissible value, the centrifugal pitch lock slide valve 8, having quite high mass, begins to move in the direction away from the axis of rotation under the action of the centrifugal force. In this case the hydraulic pitch lock passage is connected with the return passage, as a result of which the oil pressure ahead of the hydraulic pitch lock plunger decreases and valve 3 opens. As in the preceding case, after the valve closes, the propeller pitch reduction terminates and therefore the engine rpm does not increase.

The mechanical pitch lock duplicates the operation of the hydraulic lock valve. If there is oil pressure in chamber D, the spring sleeve 2 is in the extreme left position. In this case its face splines are disengaged from the face splines of the rotating sleeve 1. Mechanical connection between these sleeves is provided by means of the splined sleeve 4. If the oil pressure in the hydraulic pitch lock passage decreases, the face splines of the sleeves 1 and 5 engage with one another. At low and intermediate pitch angles (for example, less than 45°) the left end of the splined sleeve bottoms on the propeller cylinder so that movement of the rotating sleeve connected with the piston terminates. As a result of this, further reduction of the blade pitch angle becomes impossible.

Intermediate Blade Stop

To prevent the blades from going to ϕ_{min} in flight if there is an engine failure or if the engine is throttled back significantly, the blades are set on the intermediate stop. The magnitude of the angle ϕ_{is} is usually chosen to ensure conditions for safe landing of the airplane.

During the landing approach the engine is throttled and the propeller blades tend to go to the position ϕ_{min}. The result is the appearance of negative thrust, which makes landing of the airplane much more difficult. Moreover, when the propeller blades go to ϕ_{min} the engine acceleration capability deteriorates. On this basis the propeller design provides for locking the blades on an intermediate stop, to which they are usually set when the engines are throttled back severely and in emergency cases when the engine is shut down. The magnitude of the intermediate stop angle is in the 10-25° range for the various TPE types.

This device operates as follows. As the propeller pitch decreases, the piston 6 moves to the left, forcing oil from the cylinder front chamber into the engine case. When the piston reaches the position in which passage d opens, blocked prior to this time by the piston wall, the pitch lock valve 3 closes, since oil is returned through passage d to the engine case, and therefore the oil pressure on the end of the plunger decreases.

After closing of the pitch lock valve, further decrease of the propeller pitch terminates. The airplane completes the landing with this blade pitch angle, at which the propeller usually develops a small positive thrust. After touchdown the pilot throws special switches to take the propeller blades off the intermediate stop. The pitch lock valve opens and the propeller blades go to ϕ_{min}. In this case negative propeller thrust is created and is used to shorten the airplane landing roll.

In case of engine failure (shutdown) in flight, the intermediate blade stop (see Figure 197) limits the negative thrust overshoot (in comparison with ϕ_{min}).

The angle of the intermediate stop (blade low pitch angle limiter), selected to ensure safe airplane landing, cannot completely prevent or even reduce markedly the negative thrust in flight at high speeds. Therefore the need arises to use a variable intermediate stop which alters its position as a function of the flight speed. At a given altitude the variable stop must provide higher blade intermediate stop angles with increase of the flight speed, and lower angles with decrease of the speed.

Feathering

The most radical technique for preventing the occurrence of large negative thrust or propeller overspeeding in flight is to set the blades to the feathered position ϕ_{feath}, in which they have minimal drag (Figure 201). The blades are moved to the feathered position when the engine is shut down or in case of problems which might lead to the occurrence of large negative thrust or overspeeding of the propeller on an operating engine. The feathering systems of modern airplanes are made with manual and automatic control.

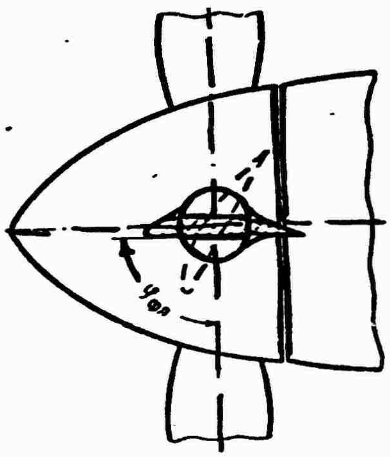


Figure 201. Blade in feathered position.

The blades are moved automatically to the feathered position if engine torque decreases during airplane takeoff (takeoff autofeather), if negative thrust of a definite magnitude occurs in any flight regime (all-regime autofeather), or if the limiting permissible engine rpm is exceeded.

We shall examine a simplified propeller feathering system schematic (Figure 202). In addition to the governing system elements which we are already familiar with, this system includes the feathering pump 3, driven by the electric motor 4; airplane oil tank 1, with a special propeller feathering reservoir; fuel controller 6 with limiting permissible engine rpm feathering sensor 7; torque feathering sensor 8; negative thrust feathering sensor 2; and other control components.

Feathering by <u>manual activation</u> of the system is usually used for training purposes or in case of failure of the automatic control components. In these cases the pilot depresses the control button 9, closing contacts b and d (as shown in the schematic). After this,

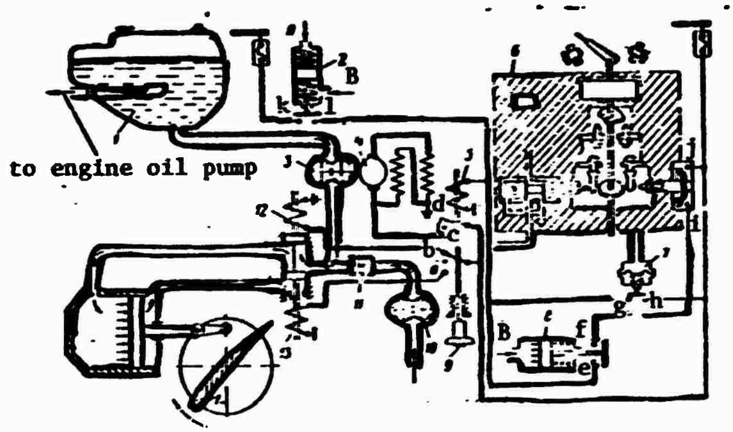


Figure 202. Propeller feathering system: 1 - tank; 2 - negative-thrust automatic feathering sensor; 3 - feathering pump; 4 - electric motor; 5 - automatic feathering system activation solenoid; 6 - fuel controller; 7 - overspeed automatic feathering sensor; 8 - torque-sensitive automatic feathering sensor; 9 - manual feathering/unfeathering button; 10 - governor pump; 11 - check valve; 12 - propeller feathering slide valve solenoid; 13 - propeller unfeathering slide valve solenoid; A - from fuel controller; B - from negative-thrust feathering sensor; C - from torquemeter.

current from the ship's system is supplied to the oil pump electric motor and the solenoid 12 of the feathering slide valve. After the electric circuit is made, the solenoid displaces the slide valve to the upper position, and the electric motor and pump supply oil from the airplane oil tank to the propeller cylinder group. The piston then moves to the right to the stop ϕ_{feath}, and the blades go to the feathered position. On actual propellers a hydraulic valve is used in place of the electro-hydraulic valve, and to feather the blades it is only necessary to depress the button momentarily, after which it is held in this position by a special electromagnetic device. At the same time the feathering pump is activated, a signal is sent to shut down the engine by closing off the fuel supply. After the blades reach the feathered position, the electrical system is automatically deactivated by a time delay relay.

Feathering by the automatic system using the torque sensor is provided in case of engine failure in the takeoff (or nearly takeoff) power regime. At takeoff, when the airplane flight speed is still low, the occurrence of negative thrust on even one of the engines cannot be tolerated, since this can lead to a crash. Consequently, the sensor which reacts to the appearance of negative thrust cannot be used.

We know from the engine design course that TPE torque is measured by the magnitude of the oil pressure in a special meter (torquemeter). Decrease of the torque (oil pressure in the torquemeter) with the engine power lever in a fixed position during airplane takeoff can be caused only by engine failure. The sensor 8, which reacts to change of the engine torque, is used to activate the feathering system and therefore prevent reduction of the blade pitch angle to values at which negative thrust develops. The sensor is triggered at the moment the oil pressure in it reaches the preset minimum corresponding to some positive thrust.

In order that this sensor not operate at low engine power settings (during throttling), its electrical system is interlocked with the fuel controller. This means that the torque sensor can activate the feathering system only if the fuel controller arm corresponds to a high engine power setting, i.e., when contacts i and j are closed.

Let us assume that the engine torque and therefore the oil pressure in the torquemeter system decrease during airplane takeoff and reach the minimal value at which the feathering system is activated. When the oil pressure transmitted from the torquemeter system to the sensor 8 decreases, its piston moves to the left under the action of the spring, closing contacts e and f. Since the fuel controller contacts i and j are closed in the takeoff regime, current from the ship's system flows to solenoid 5 and energizes the oil pump electric motor and the slide valve solenoid 12. The further operation of the feathering system is accomplished similarly to the case described above for the manual feathering case.

The propeller blades are driven to the feathered position by the automatic system using the negative thrust sensor in case of engine failure from any operating regime when the negative thrust reaches a definite (specified) value. In this connection negative thrust feathering is usually termed all-regime feathering.

The operation of the negative thrust automatic feathering system proceeds as follows. In normal flight the propeller develops positive thrust and the propeller shaft, being shifted forward, permits oil to enter the oil passage of the sensor 2. The position of the auto-feather valve components shown in Figure 202 corresponds to this propeller operating regime with positive thrust. The oil pressure from the sensor oil passage and the lower spring force act on the piston from below, and the oil pressure created by the fuel controller pump and the upper spring force act from above. In this case the force balance is such that the piston is in a slightly raised position and contacts k and l are open.

When negative thrust develops, the propeller shaft shifts aft (within the limits of the established clearances), opening the valve which returns oil from the feathering system oil passage into the engine case. The oil pressure in the chamber below the piston of the autofeather sensor 2 decreases, as a result of which the piston moves downward and closes contacts k and l. Current from the ship's system is applied to solenoid 5, which actuates and energizes the oil pump electric motor and the solenoid 12. Thus the feathering system is activated automatically and the propeller blades begin to move to the feathered position.

The propeller is driven into the feathered position by the automatic system based on maximal permissible engine rpm when for one reason or another the rpm increases above the established limits. The sensor used is a pressure switch connected to the oil system of the fuel controller, in which the pressure changes as a function of the oil pump (engine) rotational speed. With increase of the fuel controller pump rotational speed the oil pressure increases and vice versa. When the rotational speed increases above the specified limit, the oil pressure increases to the degree that the sensor membrane 7 deflects downward, overcoming the spring force, and closes contacts g and h. After this the ship's system current, as in the preceding cases, is applied to solenoids 5 and 12 and activates the automatic feathering system.

In addition to the propeller feathering methods listed above, in actual construction provision is also made for manual emergency systems which utilize the energy of the airplane's hydraulic or pneumatic systems. These systems are used in emergency cases, when the electrical system is out of order or deenergized and the solenoid 12 cannot lift the slide valve which controls entry of the propeller into the feathering position. When the emergency system is activated, gas or liquid pressure lifts the propeller feathering slide valve up, opening access of oil from the governor pump into the propeller cylinder group. Since the engine and governor pump rotational speed decreases as the propeller blades approach ϕ_{feath}, the oil pressure falls off. Therefore only partial (incomplete) feathering may be obtained.

Unfeathering of the blades, both on the ground and in the air, is accomplished manually using the same system, by pulling out the button 9. In this case contacts a and c are closed and current from the ship's system flows to the oil pump electric motor and the solenoid 13. The latter moves the slide valve downward. The oil supplied by the feathering pump enters the aft chamber of the cylinder and the oil from the front chamber drains into the engine case.

Propeller unfeathering on the ground is usually performed until the blades reach ϕ_{min}, while in flight the magnitude of the angle to which the blades must be set is determined by the autorotation rpm required to start the engine.

Construction

Blades

With regard to the number of blades, propellers are subdivided into two-bladed and multi-bladed. In selecting a propeller for an airplane the number of blades is determined by the engine characteristics (power, design flight altitude, distance between nacelle and fuselage or between nacelles). Experimental studies show that a six-bladed propeller absorbs 180-185% of the power absorbed by a three-bladed propeller of the same diameter. However, propeller efficiency decreases with increase of the number of blades, since in this case the blades operate in more disturbed flow than do the propellers with a smaller number of blades.

In this connection, on high-power engines, where the four-bladed propeller does not have adequately high efficiency, a coaxial propeller (Figure 203) can be installed rather than increasing the number of blades. The coaxial propeller consists of a pair of propellers positioned one behind the other and rotating in opposite directions about the same axis. The coaxial propellers have the following advantages:

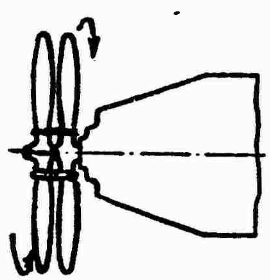


Figure 203. Coaxial propeller.

rotation of the propellers in opposite directions straightens the slipstream, which increases the efficiency of the entire installation;

the straightened slipstream flows symmetrically over the airplane lifting and control surfaces; therefore the usual airplane tendencies to bank and yaw do not arise, which are particularly undesirable at low speeds when controllability is poor;

the overall reactive moment of two identical coaxial propellers is zero, so there is no need for large trim tabs and the airplane flight characteristics are improved;

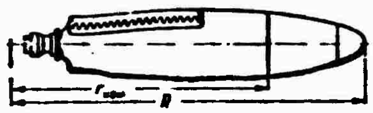


Figure 204. Propeller blade with electric heating.

the overall gyroscopic moment in curved flight is also zero, which improves airplane maneuverability;

the propeller diameter is less than when a single propeller is used.

The blades for modern propellers are made from Dural (alloy D1). The propeller blank is usually forged and then subjected to machining and heat treatment and finally painted.

The propeller blade (Figure 204) has a profiled portion called the tongue and an unprofiled part called the shank. The tongue is characterized by the relative values of the width and thickness of the sections and by their distribution along the blade radius, and also by the blade twist.

The relative width of the blade section is measured in fractions of the diameter D

$$\bar{b}=\frac{b}{D}.$$

For the modern propellers the maximal relative blade width amounts to 8-10% (Figure 205). The distribution of the relative blade width along the propeller radius characterizes the blade planform. Propeller thrust increases as the maximal value of the width approaches the blade tip. However, this increases the bending moment considerably because of center-of-pressure movement toward the blade tip. The activity factor

$$\Phi=\frac{10^5}{16}\int_{\bar{r}_0}^{1}\bar{r}^3\bar{b}d\bar{r}.$$

is used to evaluate blade planform effectiveness. Modern propellers have an activity factor of 90-110.

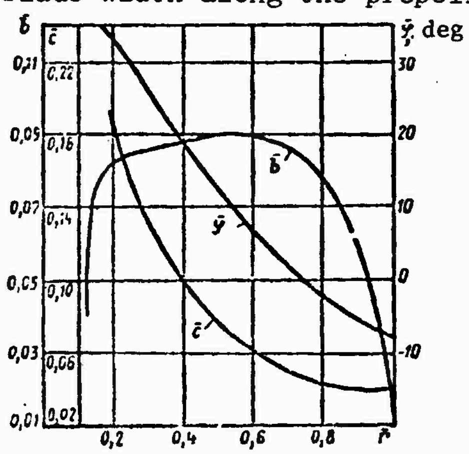


Figure 205. Blade geometric characteristics.

The relative blade section thickness is measured in fractions of the width $\bar{c}=\frac{c}{b}$. The relative thickness decreases gradually toward the blade tip. To evaluate the blade thickness we usually take its value at $\bar{r}$ = 0.9 and denote it by $\bar{c}_{0.9}$. For modern propellers $\bar{c}_{0.9}$ = = 4-5%.

The cylindrical shank part of the blades has a special thread for retention in the sleeve. This makes it possible to replace blades under base maintenance shop conditions, change the blade pitch angle when necessary, and also makesit possible to transport the propellers with the blades unscrewed without requiring that the hub be disassembled.

For propellers having left-hand rotation (looking forward) the blades under the influence of the centrifugal force transverse component moment tend to rotate counterclockwise in the sleeve thread. The thread on the shank is a left-hand thread in order to ensure that the blade will always tend to screw into the sleeve under the influence of this moment. Propellers with right-hand rotation have a right-hand thread on the blade shanks.

To fix the blade relative to the sleeve, a clamp 2 (Figure 206) is installed in a groove in the blade and clamps the split part of the sleeve 1 to the cylindrical surface blade 6. Moreover, in some propellers the blade is additionally retained in the sleeve by means of the pin 3. Centering of the blade relative to the sleeve is achieved by two cylindrical surfaces of different diameter on the shank part. On the cylindrical surface of smaller diameter there is pressed the special ring 7, which protects the butt against work hardening which can arise from blade vibration. The shank is drilled out to some depth

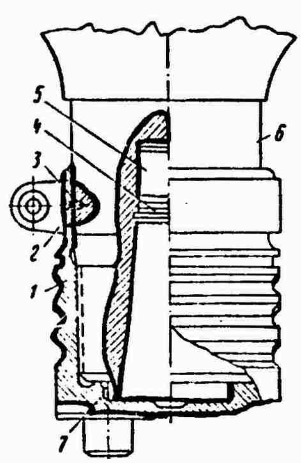


Figure 206. Blade retention in sleeve by clamp.

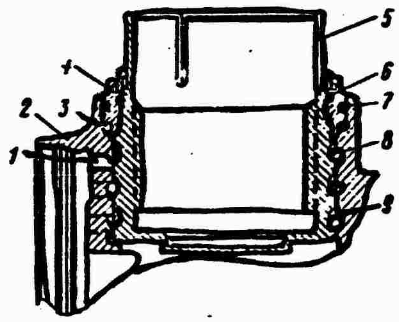


Figure 207. Sleeve in propeller hub barrel.

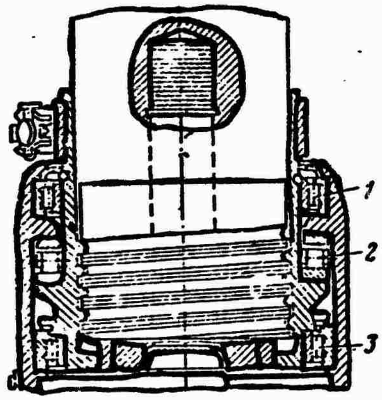


Figure 208. Sleeve with roller bearings.

to provide space to install the balancing weight 5 and washers 4.

The steel sleeve 5 (Figure 207) is an intermediate part between the blade and the propeller hub (barrel). On the outer surface of the sleeve there are grooves 9 which serve as races for the balls 8. Thus the balls, located in the races between the sleeve and the propeller hub 2, form a multirow radial thrust bearing which permits the sleeve together with the blade to rotate freely relative to the barrel. Three special drilled passages 1 are provided in the hub for filling the races with the balls. After the grooves are filled with the balls the sleeve is tightened by the nut 4, which has inner 6 and outer 7 seals and the thrust balls 3. As this is done all three rows of balls are shifted in the direction of the nut 4 and they cannot fall out through the assembly grooves.

Figure 208 shows a sleeve with three roller bearings. The middle bearing 2 is a thrust bearing which takes the centrifugal force loads, while the upper 1 and lower 3 bearings take the bending moments and are therefore radial bearings. This type of sleeve can be installed in a split hub.

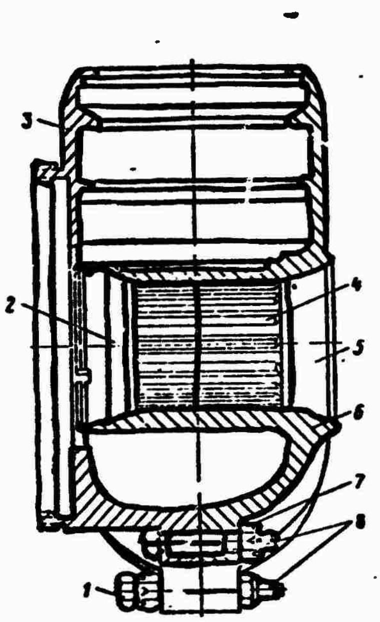


Figure 209. Split three-bladed propeller hub: 1 - balance nut; 2,5 - conical seats for front and rear cones; 3,6 - front and aft hub sections; 4 - splines; 7 - nut; 8 - through-bolts.

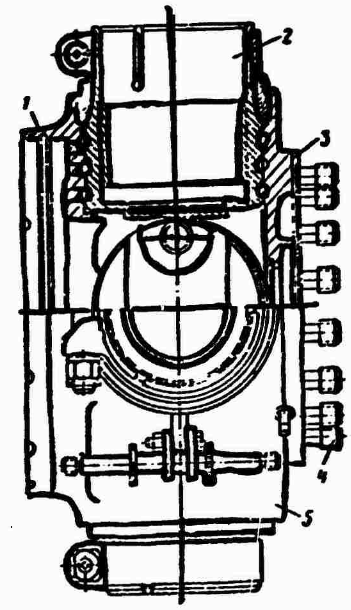


Figure 210. One-piece four-bladed propeller hub: 1 - annular shoulder; 2 - sleeve; 3 - radial splines; 4 - stud; 5 - socket (barrel).

Hub

The propeller hub serves for attachment of all the parts and components, and also for mounting and retaining the propeller on the engine reduction gearing shaft. The hub takes all the loads which develop on the blades and in the cylinder group.

Depending on the propeller type, the hub may be split (Figure 209) or one-piece. The two parts 3 and 6 are joined by the bolts 8. The hubs of modern propellers are usually one-piece. The blades are retained in the hub by sockets (barrels) 5 (Figure 210) in which the sleeves 2 are located. An annular shoulder with internal thread is provided in the front part of the hub for attaching the propeller cylinder by means of a special nut.

The propellers can be mounted on the engine in two ways: using special flanges connected by studs, and by a nut on the splined engine reduction gearing shaft.

In the first case, there are radial splines on the aft face of the hub to take the engine torque, and threaded holes into which the attaching studs 4 are screwed. On the engine flange there are mating radial splines and holes for the studs. The latter take the propeller thrust.

In the second case the hub is hollow with internal splines 4 (see Figure 209) and conical seats for the rear and front centering cones. The contact rings of the anti-icing system are attached to the aft part of the hub by special bolts.

The coaxial propeller hub (Figure 211) consists of two independent units. These propellers operate together with a governor assembly (each propeller has its own governor). The hydraulic mechanism for varying the pitch of these propellers is the two-way-action type. The blades are turned in the direction of increasing pitch under the action of oil pressure created by the governor pumps in chambers A and by the moments of the aerodynamic forces, while they are turned in the direction of decreasing pitch under the action of the oil pressure created by the engine pumps in chambers B and by the moments of the blade centrifugal force transverse components.

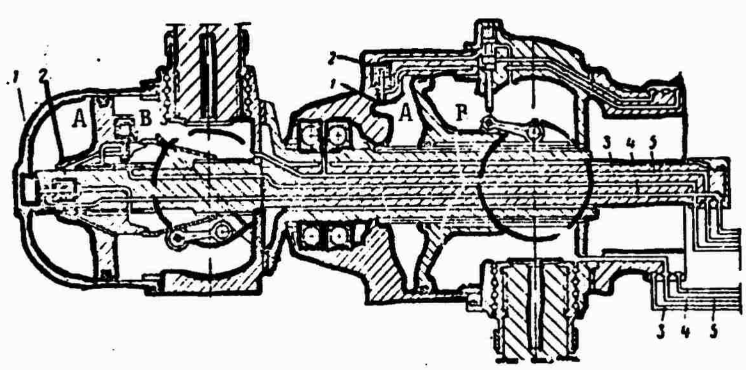


Figure 211. Coaxial propeller hub: 1 - pitch lock valve; 2 - plunger; 3,4,5 - lock, high-pitch, and low pitch valve passages.

The one-piece front propeller hub does not differ in principle from the hub of the single propeller. The peculiarities of the aft propeller hub result from the presence of the front propeller shaft and the case, which is bolted to the front face of this hub. The case is made from dural and is a structural part of the propeller. It takes the weight and thrust of the front propeller and transmits them through the aft propeller hub to the engine shaft. A groove for a ball bearing is provided in the front part of the case. In the body of the case there are oil passages and a recess for installing the pitch lock.

Cylinder Group

The cylinder group of the nonfeathering propeller intended for installation on the piston engines of the older types of airplanes is usually a simple device consisting of the cylinder 1, piston 3, and oil transfer line 2 (Figure 212). As a rule the modern propellers are of the feathering type and are more complex in construction. In addition to the cylinder, the cylinder group includes a piston group, oil transfer lines, and various protective devices.

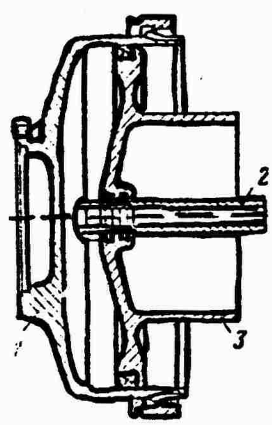


Figure 212. Cylinder group.

The cylinder is one of the structural components of the propeller. It is made from Dural and is heat treated. The cylinder is mounted on the front face of the propeller hub by means of a nut, and the sleeve used to center the propeller hub spinner is mounted on the front face of the cylinder itself. The piston group is located inside the cylinder.

Operation

Propeller Assembly

As a rule propellers are transported from the manufacturer's plant to the operational units in a semi-assembled condition (with the blades removed). The hubs and blades are subjected to external preservation prior to packing in special boxes. The propeller can be transported in assembled form to the operational unit only from an overhaul facility located close to the airfield. In this case the propeller is transported on a special cart.

Prior to installing the propeller on the airplane, the external preservative is removed from the hub and blades by means of a cloth soaked in gasoline or kerosene. The threaded (shank) parts of the blades and the internal surfaces of the sleeve are wiped off particularly carefully. After wiping they must be dried in order to prevent shifting of the blades after the sleeves are clamped up.

When mounting the blades in the propeller hub, we must remember that this operation is performed in strict accordance with the assembly marks stamped on the blades, tightening clamps, and hub sleeves. Proper assembly ensures that the factory balancing of the propeller will be maintained. We must also make certain that the blade lengths and pitch angles are identical after the blades are finally screwed into place. The blade clamp nuts are tightened with a calibrated torque wrench to a torque of 40-45 kgf·m. After performing these operations the wires of the blade heating elements are usually connected to the slipring terminals and the propeller spinner disks are installed.

It is recommended that the entire assembly of the propeller, except for the spinner disks, be accomplished on a special table. The spinner disks are installed with the propeller raised slightly. In this operation care is taken to ensure that the disks are perpendicular to the propeller axis. Otherwise the spinner will wobble and can cause powerplant vibration. Spinner wobble is eliminated by placing shims under the nut.

Before installing the propeller on the engine shaft a check is made to insure that the seals in the system for supplying oil to the propeller are in place and in good condition, after which the oil transfer line and the other components of the oil system in the shaft nose are connected. The condition of the face splines on the hub and engine and the threads on the propeller mounting bolts and nuts are checked. The splines and threads are carefully cleaned with kerosene before inspection. Damage to or dirt on the face splines can lead to unacceptable wobbling (misalignment) of the propeller, which will cause engine vibration. After cleaning, the bolt threads are lubricated with oil and checked for freedom of motion by screwing the nut on by hand.

A three-ton crane is usually used to hoist the propeller. It is recommended that the propeller be lifted and moved to the engine slowly, holding the blades by hand to avoid damage to the oil system components in the nose of the engine shaft. Special care is taken to ensure that the propeller and engine shaft are aligned — even very small misalignments cannot be tolerated. In tightening the nuts after seating the propeller in place it is very important to ensure that they are tightened uniformly. The nuts are finally tightened with a calibrated torque wrench to a torque of 18-20 kgf·m.

If the propeller is mounted on a stub shaft rather than on a flange, centering of the propeller is achieved with the aid of rear and front cones, and the propeller is restrained by tightening the nut. This type of propeller is mounted on the stub shaft with the cylinder group removed.

Prior to installing the propeller, a check is made of the condition of the centering cones, the longitudinal splines on the shaft and in the propeller hub, and the threads of the retention nut and the stub shaft. All these components must be cleaned with kerosene and lubricated with clean oil or technical-grade vaseline. No mechanical damage of these parts can be tolerated. After the nut is installed on the stub shaft it is tightened to a torque of 100-140 kgf·m (in accordance with the instructions for the given propeller), and then the oil system components and the cylinder group are installed.

Regardless of the propeller type and method of attachment, after completing all the assembly operations the propeller tracking is checked. The amount of out-of-track is measured at the blade control sections (marked with paint). To do this a table (ladder) is placed at the rear of the propeller blades and calipers are used to measure the deviation from some selected reference point to the control section of the blades as the propeller is rotated. Blade out-of-track tolerances are established for each propeller type. After this check the spinner is installed on the hub.

Checking Operation of Propeller and Governor

Prior to each flight, the operation of the propeller and governor is checked with the engine running.

The propeller and governor of the piston engine are checked when making the engine test run The entire complex of test operations performed in checking the engine is shown in Figure 213. We shall examine only the segments relating to checking the operation of the propeller and governor (shown by the solid lines).

After the engine is started and warmed up it is advanced to rated power and its operation is checked both by means of the monitoring instruments and by ear. In this check the blades must be set to ϕ_{min} (propeller control lever forward); otherwise the engine may not develop rated rpm. When checking the ignition system the blades are also on the ϕ_{min} stop. The ignition system is checked alternately on each magneto, and proper operation is judged by the magnitude of the rpm drop. If in this operation the propeller is even partially loaded the results of the ignition system operational check will be invalid,

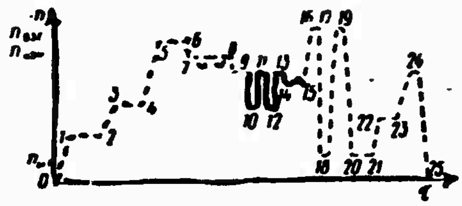


Figure 213. Piston engine test run diagram: 0-1 - start; 1-2, 3-4 - warmup; 5-6 - check at rated power; 7-8 - check of ignition system; 9-13 - exercising propeller; 14-15 - checking propeller operation at equilibrium rpm; 16-17 - takeoff power check; 17-20 - acceleration check; 20-21 - idle power check; 22-23 - cooldown; 23-24 - plug burnoff; 24-25 - shutdown.

since the rpm drop resulting from a problem in the ignition system will be compensated by decrease of the propeller pitch angle.

After this the operation of the propeller and governor is checked. The propeller is first exercised from low to high pitch and back two or three times by moving the propeller control lever aft and forward with the throttle in a fixed position (segments 9-13). The engine rpm must follow the propeller lever position change, i.e., when pulling the lever back, the rpm must decrease and when moving the lever forward the rpm must increase. This operation warms up the oil in the cylinder group by replacement of the oil two or three times, which ensures more reliable operation of the governing system. Overspeeding of the propeller during takeoff is possible if the oil in the cylinder group is not first warmed up, particularly in the winter, since cold oil has high viscosity. This makes the propeller-governor system less sensitive during takeoff as the engine rpm increases with change of the airplane flight speed.

The propeller is run from low pitch to high pitch and back at a manifold pressure somewhat less than rated-power manifold pressure. Loading up the propeller with the throttle fully open is not recommended, since detonation is possible in this case.

After warming up the cylinder group the operation of the propeller and governor is checked at an equilibrium (set) rpm. To accomplish this the propeller lever is moved aft to establish some (set) rpm (segment 14), and then the throttle is smoothly retarded and advanced. If the governing system were perfect, the rpm would remain constant as the throttle is moved. In real systems there is some lag, as a result of which rpm overshoots appear, i.e., when reducing the manifold pressure the rpm decreases somewhat from the set value and then recovers; conversely, when the manifold pressure is increased the rpm increases, with subsequent recovery (segments 14-15). After finishing these checks of the governor and propeller, the latter is set to low pitch for further checking of the engine.

Let us examine in the same sequence the procedure for checking the operation of the propeller and governor on a turboprop engine. The start is made at the minimal blade pitch angle. Starting the turboprop engine at blade pitch angles greater than ϕ_{min} (for example, at the intermediate stop angle) is not permitted, since the engine will overheat and may not start. This is explained by the increase of the torque required to rotate the propeller, and the starter power is not designed for this load.

After starting, the engine is warmed up at idle rpm with the blades remaining at ϕ_{min}. The sequence and extent of the checks made in testing the engine are determined by the operating instructions for the given engine type.

For the TPE with a governing program following the law n = const, the check of the propeller and governor operation is shown on the engine test chart (Figure 214, the triangle means that the blades are on the intermediate stop — the circle means that they are off the stop). Warmup of the cylinder group and the propeller-governor oil system is accomplished by exercising the propeller blades two or three times by varying the engine operating regime from idle to 60-70% of rated power and back (segment 3). Thus, on the TPE the blades are changed from low to high pitch and back by varying the power lever position.

The engine is warmed up at point a (Figure 215). Increase of the engine operating regime as the power lever is advanced takes place along the line ab. This is accompanied by increase of the rpm and

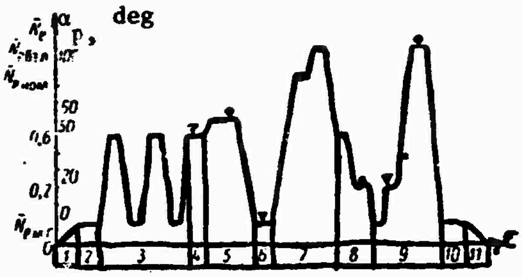


Figure 214. TPE test run diagram: 1 - start; 2 - warmup; 3 - warmup oil in cylinder group; 4 - check partial feathering; 5 - check torque-sensitive feathering; 6 - check negative-thrust feathering; 7 - check at rated and takeoff power; 8 - check intermediate stop; 9 - check acceleration; 10 - cooldown; 11 - shutdown.

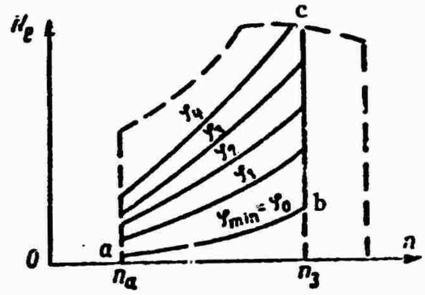


Figure 215. Engine power variation with variable and constant rotor rpm.

engine power while maintaining ϕ_{min}. After the rpm reaches the set value n_s, further advance of the power lever causes increase of the engine power and increase of the blade pitch angle at constant rpm. The extreme forward power lever position corresponds to point c.

It is obvious that when the engine operating regime is reduced (aft movement of the power lever) the engine power and blade pitch angle decrease at n = const (displacement along the vertical cb). When the fuel supply to the engine decreases to the point that the blade pitch angle reaches ϕ_{min}, further aft movement of the power lever causes engine rpm and power to reduce at ϕ_{min} = const (displacement along the curve ba). Consequently, forward and aft movement of the power lever on the segment bc causes change of the blade pitch angle and change of the oil in the cylinder group.

In addition to the propeller and governor operation checks described above, prior to every flight a check is made of the operation of the feathering system. The extent and sequence of this operation is defined by the operating instructions for the given engine type. For most engines the feathering system is checked:

from the torquemeter sensor;

from the negative thrust sensor;

by partial manual feathering of the propeller with the engine operating or stopped.

Automatic feathering of the propeller from the torquemeter sensor is checked on the ground with the engine operating. To do this the engine is advanced nearly to rated power, the system test switch is activated, and then the power lever is retarded to idle (i.e., engine failure is simulated). Activation of the test switch locks out the engine shutdown system and makes it possible for the propeller blades to move to the minimal pitch angle.

Proper operation of the torquemeter automatic feathering system is monitored by illumination of the warning lights and stable operation of the engine at idle power. Thus the torquemeter sensor feathering system is checked by means of artificial techniques which prevent engine shutdown and make it possible to evaluate the operation of the individual system components.

To check the negative thrust feathering system, a special valve is provided whose activation simulates the occurrence of negative thrust and also lowers the pressure in the line supplying oil to the all-regime autofeather sensor. In this case automatic operation of the feathering system is activated just as it would be if negative thrust were to occur.

Check of system operation by partial feathering can be accomplished with the engine either running or not running by depressing the partial feathering button. On the operating engine this feathering system check is made at 50-60% rated power. Change of the engine rpm is observed on the instruments as the partial feathering button is briefly depressed. When the rpm has decreased by 200-300 rpm, the button is released and the rpm recovers to the equilibrium value. When the button is depressed, the light indicating operation of the feathering pump will be on. With the engine not running, proper operation of the system is checked by rotation of the blades and lighting of the monitor lamp when the button is depressed.

Airstarting the Turboprop Engine

Starting the TPE in flight after the blades are in the feathered position is usually performed only for test purposes or for flight crew training.

Considering that reliable engine restart in flight is possible only in a definite range of flight altitudes and speeds, in case of engine shutdown at high altitude the pilot must descend to an altitude at which the engine will restart reliably and establish the flight speed specified in the flight operating handbook for the given airplane type. To accomplish the start the ignition is first turned on and then the propeller feathering button is pulled out. When the propeller begins to autorotate, fuel supply is initiated. The engine is usually started with the intermediate propeller stop switch in the "Prop on Stop" position. During the start the blade pitch angles change from the feathered position to the intermediate stop. When the engine rpm increases as a result of the operating turbine to the speed at which the governor is set, the propeller pitch increases. In this case the blades go to the angle corresponding to the given flight condition (flight speed and engine power).

During the engine start, there is a sharp negative thrust increase as the pitch angle decreases. The negative thrust reaches its maximal value when the blades are on the intermediate stop, with the turbine operating and the rpm approaching the governor setting. For this reason restarting the engine after shutdown and feathering the propeller is not permitted in normal flight operations. If the flight crew is not adequately trained, the abrupt negative thrust increase can lead to an accident.

To reduce the negative thrust it is desirable that the engine be started at a low flight speed. If the restart is successful, the negative thrust acts for only a short period, and a trained flight crew can handle the airplane under these conditions without any particular problem. However, if the engine restart in flight is not a good start (the engine "hangs up") and the engine does not transition to the desired operating condition, large negative thrust will act for a long time period and will cause considerable difficulty in handling the airplane. In this case the restart must be terminated and the blades refeathered.

Propeller Maintenance

Careful propeller maintenance is required during operation. This is explained by the complexity of the propeller construction and the severe operating conditions. We need only point out that the blade centrifugal forces tending to tear them out of the hubs reach 50 tons or more for certain types of propellers.

After each flight the blades are wiped with a cloth and carefully examined. Particular attention is devoted to the condition of the parts which retain the blades in the hub, and especially the condition of the blade surfaces, which are subject to mechanical damage (dents, nicks, cracks). In most cases the reason for damage of this type is poor condition of the taxiways, runways, and parking ramps, which leads to pieces of concrete, slag, or small stones striking the propeller when the engine is running. The primary technique for protecting the propeller against such damage is strict adherence to the requirements on care of the airplane parking areas, taxiways, and runways. These areas must be clean and the paving must be undamaged.

It is recommended that minor mechanical damage to the blades be repaired without removing the propeller from the airplane. Elimination of these defects excludes the possibility of the appearance of stress concentrations, which reduce blade strength, and also restores blade aerodynamic efficiency.

The following blade defects can be repaired in the field.

1. Nicks in the leading and trailing edges of depth no more than 3 mm. They are eliminated by fairing them out using contoured files with smooth transition over a distance of 20-30 mm along the edge and the same sort of smooth rounding along the contour (profile).

After filing, these areas are polished using No. 200 emery cloth. Damaged areas on the leading edge must be handled particularly carefully in order not to disrupt the general thrust distribution pattern along the blade profile and thereby avoid reduction of the propeller efficiency.

2. Scratches, score marks, dents, and nicks on the blade face no more than 0.5 mm deep. These are removed using a scraper with subsequent polishing using No. 200 emery cloth. In so doing a smooth transition to the undamaged blade surface must be maintained.

3. Nicks and tears at the blade tips of depth no more than 10 mm. These defects are filed out using a suitable template with subsequent polishing. This same template is used to file off the tip of the opposite blade to prevent unbalancing the propeller.

Segments of the blade tongue which have been stripped must be painted after the work is completed. This is done to protect the surface against corrosion. These segments are degreased with gasoline and then coated with ALG-1 primer when an oil base enamel is to be used, or with 138-A primer when aerolacquer is to be used.

After application the primer must be dried at a temperature of 15°C for at least twenty four-hours.

More severe damage than noted above cannot be repaired under field conditions. In these cases the propeller is removed from the airplane and sent to overhaul facilities. This is also done if cracks, bending of the blade tongue, or damage to the anti-icing coating are detected.

Oil leakage from the propeller hub cannot be permitted. If oil leakage from under the propeller cylinder nut is detected, this defect is repaired under field conditions by removing the cylinder group and replacing the faulty seal.

If oil leakage from the blade sleeves occurs because of faulty rubber seals, the propeller is usually sent to an overhaul facility.

In the wintertime at low ambient air temperatures the engine, propeller cylinder group, and hub are heated with hot air from ground support heaters. To do this the propeller and engine are covered with a hood and the hot air is blown under this hood. Special valves are provided in the hood to create circulation of the hot air around the propeller hub.

CHAPTER 11

AIRCRAFT ENGINE STARTING SYSTEMS

Requirements

Engine starting is the process of accelerating the engine from the nonoperating condition to the idle power regime. Starting duration is from 30 to 120 seconds. To reduce the duration it is necessary to have sufficiently powerful starters, and this complicates the construction and increases powerplant weight.

Engine starting includes acceleration of the engine rotor, supply of fuel to the combustion chamber, ignition of the fuel, and stabilization of the engine at idle power. The starting operation requires a starter, sources of power to supply the starter, starting fuel manifolds, and ignition and control units. The entire complex of these devices and assemblies is termed the starting system.

The following basic requirements are imposed on starting systems.

1. The engine must start reliably on the ground and in flight without additional adjustment of the automatic control components or the fuel controller prior to the start. The possibility of the occurrence of flame sources which are capable of starting a fire in the airplane must be excluded.

2. Engine starting on the ground must be reliable when using either on-board or ground support equipment at ambient air temperatures from minus 50° C to plus 45° C. Equipment which facilitates the start without increasing start preparation time (for example, the use of oil preheating) may be used at ambient temperatures below freezing. The use of ground support equipment which facilitates the start but requires increased preparation time (preheating the engine and the starting system) is permitted in the development of starting systems for TJE at ambient temperatures below minus 40° C and for TPE below minus 25° C.

3. Reliable engine restarting in flight as a result of surging, flameout, or other operational abnormalities which do not cause damage to the engine components or parts.

4. Engine starting must be automated and must satisfy the following conditions:

the starting system is energized by depressing the start button;

the starting process up to engine stabilization at the specified operating conditions must be automatic, without additional manual operations being performed after depressing the starter button and moving the power lever to the starting position;

the automatic starting system ensures stable engine operation during the starting process and acceleration to the idle regime in the established time;

the system for starting the engine on the ground and in flight is automatically disengaged and prepared for the next start;

on multiengine aircraft the starting system provides the possibility of starting one of the engines and then starting the remaining engines using power from those started previously.

5. The system for starting the engines from on-board power sources must be autonomous and capable of providing a number of successive starts which is one more than the number of engines on the aircraft without intermediate recharging or refueling of the on-board equipment.

6. The engine starting system must provide for: rapid termination of the starting process, switching power for the starter from on-board to ground sources (and vice versa) without need for regulation of the system, and starting on the fuel normally used to feed the engine.

Starting Stages

Acceleration of the GTE rotor during starting is accomplished by the starter and the engine turbine, which do not both participate in the acceleration throughout the entire starting period but rather only in certain stages. In this connection the engine start can be broken down into three stages (Figure 216).

In the first stage from initiation of the start until the turbine comes into play at the rotor speed n_1, the engine is accelerated by the starter alone. In this stage the engine rotor acceleration torque is

$$M_{y_1} = M_{\text{ст}} - M_{\text{c}} = I_p \frac{d\omega}{d\tau} = 2\pi I_p \frac{dn}{d\tau} \ \text{N·m}.$$

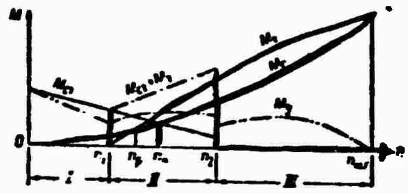
where M_{st} is the moment developed by the starter, N·m;

M_c is the moment required to rotate the compressor, accessory drive, and overcome friction, N·m;

I_p is the mass polar moment of inertia of the engine rotor (and for the TPE the propeller moment of inertia, referred to the rotor shaft), N·m;

ω is the engine rotor angular velocity, rad/sec;

n is the engine rotor speed, rps.



Figu'e 216. Engine start stages.

The following approximate formulas can te used to find the polar moments of inertia of projected engines and propellers:

the polar moment of inertia of a TPE rotor [60]

$$I_{p_p} = 0,33\, i_{\text{к}} D_{\text{к}}^4 + i_{\text{т}} D_{\text{т}}^4 \ \text{N·m·sec}^2$$

where D_{co} and D_t are the diameters of the compressor and turbine rotors, m;

i_{co} and i_t are the numbers of compressor and rotor stages;

for four-blade air propellers with Dural blades

$$I_{p_{\text{в}}} = 0,28\, D_{\text{в}}^{4,4} \quad \text{N·m·sec}^2$$

where D_p is the propeller diameter, m.

In the <u>second starting stage</u> from n_1 until the starter disengages at n_2, both the starter and turbine accelerate the rotor. In this case

$$M_{y_{II}} = M_{\text{ст}} + M_{\text{т}_{II}} - M_{\text{c}},$$

where $M_{t_{II}}$ is the moment developed by the engine turbine.

At $n = n_r$ $M_{t_{II}} = M_c$ and $M_{a_{II}} = M_{st}$.

At the rotational speed n_{co}, corresponding to the motoring speed, $M_{st} = M_c$.

In the <u>third stage</u> from n_2 to n_{idle} (to the rotor speed at idle power), the starter is disengaged and the engine rotor is accelerated by the turbine alone. In this case

$$M_{y_{III}} = M_{T_{III}} - M_c.$$

At $n = n_{idle}$ $M_{t_{III}} = M_c$ and $M_{a_{III}} = 0$.

The values of the speeds n_1, n_r, n_{co}, n_2 and n_{idle} depend on the characteristics of the compressor, turbine, and starter, operation of the combustion chamber, and configurational and operational factors (Table 8).

TABLE 8. GAS TURBINE ENGINE ROTOR SPEEDS (% OF MAXIMAL)

Engine	Compressor	Starter	$\bar{n}_1$	$\bar{n}_r$	$\bar{n}_{co}$	$\bar{n}_2$	$\bar{n}_{idle}$
TJE	Axial	Gas turbine	7-12	11-18	8-10	20-35	28-40
TJE	Two-stage axial	Electric	8-9	15-18	10-18	26-35	55-60
TJE	Centrifugal	Same	6-10	10-15	7-15	12-19	20-25
TPE	Axial	Same	8-15	17-22	10-22	30-60	50-80

Moments Acting

Engine Rotor Drag Torque

The engine rotor drag torque is made up of the compressor drag torque M_{co}, the torque required to drive the accessories, and the torque required to overcome friction. The last two torques are comparatively small (no more than 5% of the torque M_{co}), and therefore we can take

$$M_c = 1{,}05\, M_{\kappa}.$$

The compressor drag torque is proportional to the square of the rotational speed

$$M_{\kappa} = a_{\kappa} n^2.$$

Then

$$M_c \doteq 1{,}05\ M_{\text{к}} = 1{,}05\, a_{\text{к}} n^2 = an^2 \ \text{N·м},$$

where a is a coefficient of proportionality

$$a = 1{,}05\, a_{\text{к}} = \frac{M_c}{n^2}\ \frac{\text{N·m}}{(\text{rps})^2}$$

Following are the values of the coefficient a for some engine types:

Engine	VK-1	AI-20	RD-3M-500
$a \cdot 10^6$, kgf·m/(rpm)^2	5.3	2.1	180

The coefficient a for new starting system designs can be found using Figure 217. Knowing the value of the rotational speed n_{idle} at idle power, we can calculate the value of the coefficient a.

Turbine Torque

The turbine torque depends on the turbine inlet gas temperature and engine rotor speed. In the first stage of the start the turbine does not develop positive torque and acts as a brake.

The variation of the torque developed by the turbine for a constant gas temperature in the segment from n_1 (see Figure 216) to n_2 can be taken to be a linear function of the rotor speed, i.e.,

$$M_{\text{т}_{II}} = mn - p \ \ \text{N·m}$$

The values of the coefficients m and p can be found from the boundary conditions:

1) for $n = n_1$ $M_{t_{II}} = 0$ and $mn_1 = p$;
2) for $n = n_r$ $M_{t_{III}} = M_c$.

Consequently, $an_r^2 = mn_r - mn_1 = m(n_r - n_1)$.

Then

$$m = a\frac{n_p^2}{n_p - n_1} = a\frac{n_p}{1-\frac{n_1}{n_p}} \quad \frac{\text{N}\cdot\text{m}}{\text{rps}};$$

$$p = mn_1 = a\frac{n_p n_1}{1-\frac{n_1}{n_p}} \quad \text{N}\cdot\text{m}$$

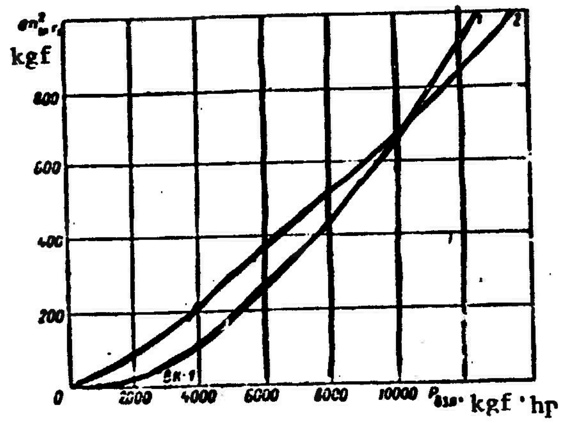


Figure 217. Values of an^2_{idle} versus TJE takeoff thrust (curve 1) or equivalent TPE power (curve 2).

On the segment from n_2 to n_{idle} the turbine inlet gas temperature is not constant and the turbine torque does not vary linearly with rotor speed. With some degree of approximation, we can take a linear dependence of the turbine torque on rotor speed in the form

$$M_{t_{III}} = m'n - p' \text{ N}\cdot\text{m}$$

The values of the coefficients m' and p' can also be found from the boundary conditions:

1) for $n = n_2$ $M_{t_{III}} = m'n_2 - p'$ and at the same time $M_{t_{III}} = mn_2 - p$.

Consequently, $m'n_2 - p' = mn_2 - p$; hence

$$p' = m'n_2 - mn_2 + p; \tag{109}$$

2) for $n = n_{idle}$ $M_{t_{III}} = M_c$.

Consequently, $m'n_{idle} - p' = an^2_{idle}$, hence

$$p' = m'n_{м.г} - an^2_{м.г}. \tag{110}$$

Equating (109) and (110), we obtain

$$m'n_2 - mn_2 + p = m'n_{\text{м.г}} - an^2_{\text{м.г}},$$

hence

$$m' = \frac{an^2_{\text{м.г}} - mn_2 + p}{n_{\text{м.г}} - n_2} \frac{\text{N}\cdot\text{m}}{\text{rps}};$$

$$p' = \frac{an^2_{\text{м.г}} - mn_2 + p}{1 - \frac{n_2}{n_{\text{м.г}}}} - an^2_{\text{м.г}} \ \text{N}\cdot\text{m}$$

The values of m, p, m', p' for some engines are as follows:

Engine	VK-1	AI-20	RD-3M-500
m, kgf·m/rpm	0.0186	0.0158	0.361
m', kgf·m/rpm	0.0178	0.0275	0.325
p, kgf·m	13.1	27.7	142.5
p', kgf·m	11.8	66.0	10

Starter Torque

Depending on the starter type, the nature of its torque variation with rotor speed may be different. For electric and gas turbine starters we can take

$$M_{\text{ст}} = M_{\text{ст}_0} - cn,$$

where M_{st_0} is the initial starting torque, N·m;

c is a coefficient which depends on the starter type, N·m/rps.

For gas turbine starters with a single turbine M_{st} = const; therefore, c = 0, while for electric and gas turbine starters with two kinematically uncoupled turbines

$$c=\frac{M_{\text{ст}_0}-M_{\text{ст}}}{n}.$$

To ensure an acceptable start duration the starter torque must exceed by 2-3 times the resistance torque at the starter rotational speed n_1

$$\frac{M_{\text{ст}_1}}{M_{\text{c}_1}}=k_1=2\div 3.$$

Hence

$$M_{\text{ст}_1}=M_{\text{c}_1}k_1=ak_1n_1^2. \tag{111}$$

Knowing the values of M_{st_1} and n_1 and assuming a value of the coefficient c, we can determine the initial starting torque for the case in which the starter torque varies linearly with rotor speed

$$M_{\text{ст}_0}=M_{\text{ст}_1}+cn_1=ak_1n_1^2+cn_1. \tag{112}$$

The maximal value of the coefficient c is fcund from the condition that the starter torque be zero at the rotor speed n_2, when the starter is disengaged

$$M_{\text{ст}_2}=M_{\text{ст}_0}-c_{\max}n_2=0.$$

Consequently

$$M_{\text{ст}_0}=c_{\max}n_2 \text{ and } c_{\max}=\frac{M_{\text{ст}_0}}{n_2}.$$

Knowing the torque M_{st_1}, the corresponding value of n_1, and the value of n_2, we can draw in Figure 218 a straight line between these two points, along which starter torque varies as a function of rotor speed. In order to make the calculations in tabular form, this relationship can be represented by the formula obtained from similar triangles

$$\frac{M_{ст_1}}{M_{ст_0}} = \frac{n_2 - n_1}{n_2},$$

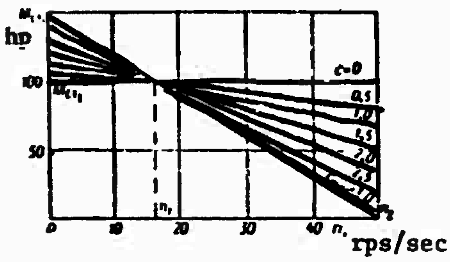


Figure 218. Starter torque versus shaft speed for given value of M_{st_1} and different values of coefficient c.

hence

$$M_{ст_0} = M_{ст_1}\frac{n_2}{n_2 - n_1} = c_{max}\, n_2$$

and, consequently,

$$c_{max} = \frac{M_{ст_1}}{n_2 - n_1}. \qquad (113)$$

As a result, the starter torque for c_{max} will be

$$M_{ст} = M_{ст_0} - c_{max}\, n = c_{max} n_2 - c_{max} n = c_{max}(n_2 - n) = M_{ст_1}\frac{n_2 - n}{n_2 - n_1}. \qquad (114)$$

We note that here and hereafter we use the concept of starter torque relative to the engine rotor shaft. If there is gearing between the starter shaft and the engine shaft with the gear ratio $i = n_{st}/n$, neglecting losses in the reduction gearing we can reduce the starter torque M_{st} relative to the engine rotor to the starter shaft torque M'_{st} on the basis of transmission and reception of the same power.

Then $M_{st}n = M'_{st}n_{st}$; hence

$$M'_{ст} = \frac{M_{ст}\, n}{n_{ст}} = \frac{M_{ст}}{i}.$$

For modern starters the gear ratio values are quite large. Thus, for gas turbine starters $i = 27 - 30$ at n_2.

In determining starter power, we can use torques relative to the engine rotor shaft without introducing the value of the gear ratio into the calculation. However, in determining the actual loads on the starter shaft and in other calculations associated with starter operation, strength, and so on, we must consider the torques on the starter shaft.

Duration of Starter Operation and Engine Start

The starter operation duration τ_{st} is made up of the starter operation duration τ_1 prior to initiation of turbine operation and the duration τ_2 of combined operation of the starter and turbine

$$\tau_{ст}=\tau_1+\tau_2.$$

The overall start duration can be determined if we add to the starter operation duration the duration τ_3 of the period of engine acceleration by the turbine

$$\tau_{зап}=\tau_{ст}+\tau_3=\tau_1+\tau_2+\tau_3.$$

Substituting the values obtained for the turbine and starter drag torques, we obtain:

for the first stage

$$\tau_1=2\pi I_p\int_0^{n_1}\frac{dn}{M_{y_I}}=2\pi I_p\int_0^{n_1}\frac{dn}{M_{ст}-M_c}=2\pi I_p\int_0^{n_1}\frac{dn}{M_{ст_0}-cn-an^2}\ \text{sec} \tag{115}$$

for the second stage

$$\tau_2=2\pi I_p\int_{n_1}^{n_2}\frac{dn}{M_{y_{II}}}=2\pi I_p\int_{n_1}^{n_2}\frac{dn}{M_{ст}+M_{\tau_{II}}-M_c}=$$
$$=2\pi I_p\int_{n_1}^{n_2}\frac{dn}{M_{ст_0}-cn+mn-p-an^2}\ \text{sec} \tag{116}$$

for the third stage

$$\tau_3=2\pi I_p\int_{n_2}^{n_{м.г}}\frac{dn}{M_{y_{III}}}=2\pi I_p\int_{n_2}^{n_{м.г}}\frac{dn}{M_{\tau_{III}}-M_c}= \tag{117}$$

(equations continued on next page)

$$= 2\pi I_p \int_{n_s}^{n_{м.г}} \frac{dn}{m'\,n - p' - an^2} \text{ sec} \qquad (117)$$

The integrals constitute an expression of the form

$$\int \frac{dn}{an^2 + cn + M_{ст_0}},$$

which makes it possible to solve (115), (116), (117) analytically. However, this solution leads to somewhat cumbersome formulas. Therefore, graphoanalytical solution can be recommended. This involves summing the magnitudes of the torques M_c, M_t, and M_{st} graphically or in tabular form and determining the accelerating torque (see Figure 216). Breaking the rotor speed range in question down into elementary segments Δn_i, we find the time values $\Delta\tau_i$ from the formula

$$\Delta\tau_i = 2\pi I_p \frac{\Delta n_i}{M_{\text{т}_{\text{ср}_i}}},$$

where $M_{a_{av_i}}$ is the average value of M_a on the segment Δn_i.

The calculated time values Δt_i are entered in the table. Summing these values, we obtain the values τ_1, τ_2, τ_3 and find starter operation duration or overall start duration.

We see from (115), (116), (117) that, other conditions being the same, the magnitude of the accelerating torque determines the start duration. The larger the accelerating torque, the shorter is the start duration.

For given values of the torques M_t and M_c, the accelerating torque can be increased by increasing the parameter characterized by the expression

$$A = \int_0^{n_2} M_{ст}\, dn = \int_0^{n_2} (M_{ст_0} - cn)\, dn =$$

$$= \int_0^{n_2} M_{ст_0}\, dn - c \int_0^{n_2} n\, dn = M_{ст_0} n_2 - c\frac{n_2^2}{2}.$$

To convert from the torque M_{st_0} to the torque M_{st} we can use (112).

Then

$$A=(M_{\text{ст}_1}+cn_1)n_2-c\frac{n_2^2}{2}=M_{\text{ст}_1}n_2-cn_2\left(\frac{n_2}{2}-n_1\right).$$

Since the quantity n_2/α is larger than n_1 (see Table 8), for given values of M_{st_1}, n_1, and n_2 the rate at which the engine is accelerated by the starter decreases with increase of the coefficient c. Therefore, the maximal start duration and the minimal starter power correspond to the maximal values of the coefficient c.

Starter Power Required

The starter power required can be found from the known values of its torque M_{st} and the engine rotor speed n

$$N_{\text{ст}}=2\pi M_{\text{ст}}n \text{ W}. \tag{118}$$

It was shown above that the start time depends on the magnitude of the starter torque and the nature of the curve of starter torque versus rotational speed. Therefore, the interrelationship between starter power required and start duration amounts to the fact that starter power must be greater for a shorter start duration and vice versa.

Graphoanalytic Determination of N_{st}

We must know the quantities n_1, n_r, n_2, n_{idle}, a and assume a value of k_1. Then we can determine the torques M_c and M_t for various rotational speeds. We use (111) to find the value of M_{st_1}, after which we use (114) to find the value of M_{st} as a function of the rotational speed n, calculate the magnitude of the accelerating torque M_a, and plot a curve of its variation with rpm (see Figure 216). After this we find the starter operation duration.

If the starter operation duration obtained corresponds to the assumed or specified value, we can use this curve of starter torque versus rpm and (118) to calculate the values of the required starter torque.

If the starter operation duration obtained does not correspond to the specified or assumed value, we must take different values of the coefficient k_1 and repeat the calculation.

Approximate Determination of N_{st}

We can assume that $M_c = 0$ in the first stage and $M_{t_{II}} - M_c = 0$ in the second stage. These two assumptions together somewhat compensate one another. Then $M_a = M_{st}$ and the starter operation duration is

$$\tau_{\text{ст}} = 2\pi I_p \int_0^{n_2} \frac{dn}{M_y} = 2\pi I_p \int_0^{n_2} \frac{dn}{M_{\text{ст}}} \text{ sec}$$

For the case in which $c \neq 0$, the starter operation duration is

$$\tau_{\text{ст}} = 2\pi I_p \int_0^{n_2} \frac{dn}{M_{\text{ст}_0} - cn} = 2\pi \frac{I_p}{c} \ln \frac{M_{\text{ст}_0}}{M_{\text{ст}_0} - cn_2} =$$
$$= 14{,}45 \frac{I_p}{c} \lg \frac{M_{\text{ст}_0}}{M_{\text{ст}_0} - cn_2} \text{ sec} \qquad (119)$$

The starter power is

$$N_{\text{ст}} = 2\pi M_{\text{ст}} n = 2\pi (M_{\text{ст}_0} n - cn^2) \text{ W}.$$

The maximal starter power and the corresponding rpm can be found by equating the derivative of the power with respect to rpm to zero

$$\frac{dN_{\text{ст}}}{dn} = 2\pi (M_{\text{ст}_0} - 2cn) = 0.$$

Therefore, the rpm corresponding to maximal starter power is

$$n_{N_{max}} = \frac{M_{ст_0}}{2c} \text{ rps/sec}$$

and the maximal starter power is

$$N_{ст_{max}} = 2\pi (M_{ст_0} n_{N_{max}} - cn^2_{N_{max}}) =$$
$$= 2\pi \left(\frac{M^2_{ст_0}}{2c} - \frac{M^2_{ст_0}}{4c} \right) = 1{,}57 \frac{M^2_{ст_0}}{c} \text{ W}. \tag{120}$$

The following sequence is recommended for approximate determination of starter power for the case in which $c \neq 0$. We must know the values of the rotational speeds n_1 and n_2, the coefficients a and k_1, and the engine rotor moment of inertia I_r. We use (111) to find the starter torque M_{st_1}. Taking several values of the coefficient c, we use (112) to calculate the value of the initial starting torque M_{st_0}. The values of the coefficient c must lie in the range from zero to a value somewhat less than c_{max} (for c_{max} the problem solution leads to an infinite value of the time, which is explained by the small values of M_{st} at rotational speeds close to n_2). Knowing the value n_2 of the starter dropout speed, we find $M_{st_0} - cn_2$ and use (119) to calculate the starter operation duration. We use (120) to find the maximal starter power.

It is convenient to perform the calculations in tabular form. As a result we can plot a curve of maximal starter power versus starter operation duration (Figure 219). Assuming a starter operation duration, we can find the maximal starter power and vice versa.

The solution of this problem can be facilitated if we assume a required starter operation time and a value of the coefficient c somewhat smaller than the maximal value indicated in (113). Then we can find from (119) the value of

$$\lg \frac{M_{ст_0}}{M_{ст_0} - cn_2} = \frac{\tau_{ст} c}{14.45 I_p}.$$

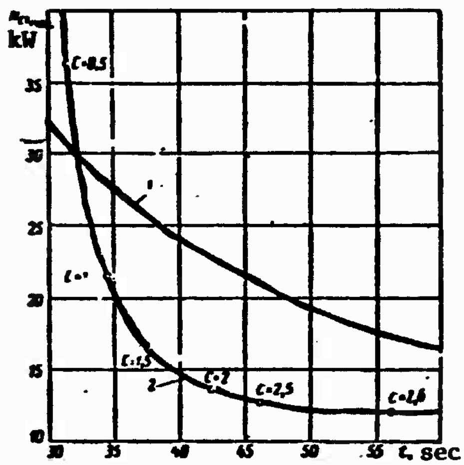


Figure 219. Starter power required versus starter operation duration and nature of torque variation: 1 – c = 0; 2 – c ≠ 0.

This makes it possible to calculate the value of M_{st_0} and from it the value of the maximal starter power using (120).

If c = 0, then $M_{st} = M_{st_0} =$ = const. In this case (119) cannot be used, since (lgl = 0). In this case, the starter operation duration can be found from the formula

$$\tau_{\text{ст}} = 2\pi I_p \int_0^{n_2} \frac{dn}{M_{\text{ст}_0}} = 2\pi \frac{I_p}{M_{\text{ст}_0}} \int_0^{n_2} dn = 2\pi \frac{I_p n_2}{M_{\text{ст}_0}},$$

hence

$$M_{\text{ст}_0} = 2\pi \frac{I_p n_2}{\tau_{\text{ст}}};$$

$$N_{\text{ст}} = 2\pi M_{\text{ст}_0} n = 4\pi^2 \frac{I_p n_2 n}{\tau_{\text{ст}}}.$$

The maximal value of the starter power will occur when $n = n_2$

$$N_{\text{ст}_{\max}} = 39{,}48 \frac{I_p n_2^2}{\tau_{\text{ст}}} \text{ W}. \tag{121}$$

The curve of maximal starter power as a function of starter operation duration obtained using (121) for the case c = 0 is shown in Figure 219. We see that with M_{st} = const far more maximal power is required than with M_{st} = var for long starter operation duration. For small values of the starter operation duration it is better to select a starter with constant value of the torque. This situation is a result of the differ nt nature of the shape of the starter torque curves (see Figure 218).

<u>Example</u>. Find the power of an electric starter for a TPE having the following specifications: $N_{eq_{max}}$ = 5000 hp = 3680 kW; n_{max} = = 10,000 rpm = 166.7 rps; I_p = 1.0 kgf·m·sec^2 = 9.81 N·m·sec^2.

From Table 8 we find n_1 = 1000 rpm = 16.7 rps; n_2 = 3000 rpm = = 50 rps; n_{idle} = 8000 rpm = 133.3 rps. From Figure 217 we find the value an^2_{idle} = 280 kgf·m = 2760 N·m. Then

$$a=\frac{an^2_{\text{м.г}}}{n^2_{\text{м.г}}}=\frac{2760}{133{,}3^2}=0{,}154\ \frac{\text{N}\cdot\text{m}}{\text{rps}^2}.$$

We take a value of the coefficient k_1 = 2.3 and use (111) to find

$$M_{\text{ст}_1}=ak_1n_1^2=0{,}154\cdot 2{,}3\cdot 16{,}7^2=100\ \text{N}\cdot\text{m}$$

We shall examine the case when the coefficient c ≠ 0. We calculate its maximal value from (113)

$$c_{max}=\frac{M_{\text{ст}}}{n_2-n_1}=\frac{100}{50-16{,}7}=3\cdot\frac{\text{N}\cdot\text{m}}{\text{rps}}$$

We take values of c from 0.5 to 2.6. From (112) we find the value of the initial starting torque. We find the starter operation duration from (119) and the maximal starter power from (120).

For the case when the coefficient c = 0,we calculate the maximal starter power using (121).

The calculations are shown in Table 9. The curve of maximal starter power versus starter operation duration is shown in Figure 219. Assuming a starter operation duration, we can find the required starter power.

Statistical data can be used for approximate maximal starter power calculations. Figure 220 shows the value of the maximal starter power of engines as a function of TJE takeoff thrust or TPE equivalent power.

TABLE 9. DETERMINING MAXIMAL STARTER POWER

c, N·m (rps)	$M_{c\pi_1}$, N·m	M_{cT_0}, N·m	$M_{c\pi_2}$, N·m	$M_{cT_0}-M_{c\pi}$, N·m	$\frac{M_{cT_0}}{M_{cT_0}-M_{c\pi}}$	$\lg\frac{M_{cT_0}}{M_{cT_0}-M_{c\pi}}$	$\frac{I_p}{c}$	t_{cT}, sec	$M^2_{cT_0}\cdot 10^{-3}$	$\frac{M^2_{cT_0}}{t_c}\cdot 10^{-3}$	$N_{cT_{max}}$, kW
0,5	8,3	108,3	25	83,3	1,30	0,11	19,62	31,2	11,6	23,2	36,4
1,0	16,7	116,7	50	66,7	1,75	0,24	9,81	34,6	13,5	13,5	21,4
1,5	25,0	125,0	75	50,0	2,50	0,40	6,54	37,6	15,6	10,4	16,3
2,0	33,3	133,3	100	33,3	4,00	0,60	4,90	42,5	17,6	8,8	13,8
2,5	41,7	141,7	125	16,7	8,45	0,93	3,43	46,2	20,0	8,0	12,6
2,6	43,5	143,5	130	13,5	10,60	1,03	3,79	56,2	20,6	7,7	12,1
0	—	—	—	—	—	—	—	30	—	—	32,2
0	—	—	—	—	—	—	—	40	—	—	24,1
0	—	—	—	—	—	—	—	50	—	—	19,4
0	—	—	—	—	—	—	—	60	—	—	16,3

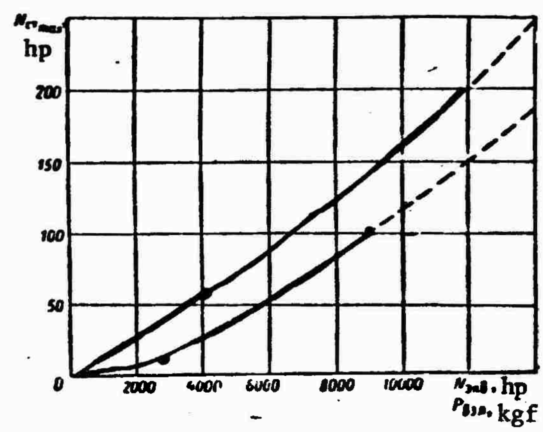


Figure 220. Required starter power versus TJE takeoff thrust (curve 1) or equivalent TPE power (curve 2).

Characteristics of Piston Engine Starting Systems

The power required to overcome rotational resistance is greater for the PE than for the GTE; however, the engine power available at low rotational speeds exceeds considerably the power required, and this permits more rapid starting. Only two or three revolutions of the crankshaft at a speed of 30 - 60 rpm are necessary to start the PE. This rotation creates the conditions for the appearance of individual bursts of power which accelerate the crankshaft to idle speed in one to two seconds.

The rotational drag torque depends on the engine design, piston friction on the cylinder wall, bearing friction, backpressure as the mixture is compressed in the cylinders, and inertia of the crankshaft and connecting rod assembly mass. The most severe starting conditions arise at low temperatures, when the oil viscosity increases. The initial starting torque is one of the basic factors affecting starting reliability. The analysis of all these factors is quite complex. The starter power for a PE amounts to about 1 - 1.5% of engine rated power.

Starters

Autonomous electric and mechanical starters are used to spin up the engine rotors (Figure 221). As a rule, these starters act on the engine rotor through a clutch, but there are air starters which feed compressed air (through a distributor) directly into the PE cylinders when the piston is still in the upper position (after passing dead center). Attempts have been made to drive the GTE rotor by directing compressed air (gas) on the turbine blades rather than through a clutch. However, the very high air (gas) consumption has prevented the introduction of this type of starting.

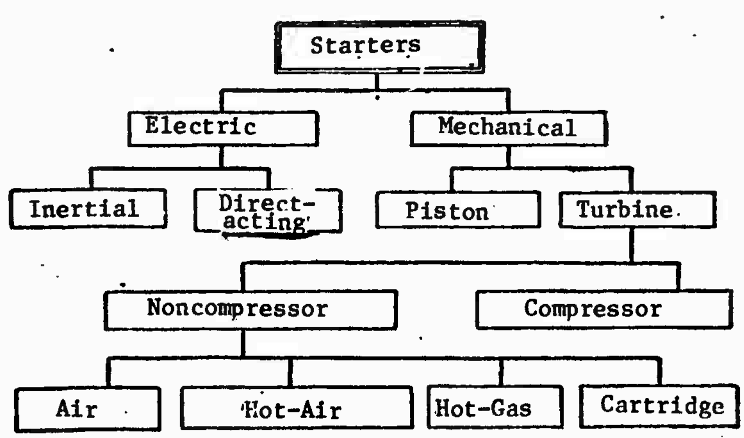


Figure 221. Starter classification.

In selecting the type of starting system,we take into account many factors, the most important being the weight, size, and starting reliability using both ground and on-board energy sources under all operating conditions.

Electric Starters

Direct-acting electric starters in which the starter is connected directly through a mechanical transmission with the engine rotor are used to start GTE. These starters are designed for short-term operation, and their power goes up to 25 kW. Inertial starters may be used to start PE. The starter flywheel is first accelerated and then the flywheel is connected with the PE crankshaft. However, the use of this type of starter for starting the GTE requires a longer motoring time, since the rotor shaft must be given a greater angular velocity than the PE shaft.

Extensive use has recently been made of starter-generators, which perform the starter function during engine starting and act as a generator after the start. This reduces the weight and cost of the starting system.

Electric starters are quite reliable in operation, simple to control, permit easy automation of the starting process, and are also simple and convenient to maintain. They are used for starting engines having comparatively low inertia or when the time required to accelerate the engine to idle speed is not very short. Increase of the starter power is required for starting engines with large moments of inertia and to shorten the time for acceleration to idle speed. Characteristic of the systems with electric starters is considerable increase of the weight when starter power is increased,which is a result of weight increase of the starters themselves and the power supplies. Under these conditions the weight characteristics of the electric systems may be inferior to those of other systems.

Direct-current electric starters are in use at the present time. Their rotational speed during the starting process is automatically regulated to follow a specified law. This law is expressed by the relation

$$\omega = \frac{U - I_a(R_a + R_{add})}{C\Phi},$$

where U is the voltage across the starter terminals;

I_a is the armature current;

R_a is the armature resistance;

R_{add} is the additional (starting) resistance in the armature circuit;

C is a constant coefficient;

Φ is the magnetic excitation flux.

We see from this formula that we can regulate the dc motor rotational speed by varying the line voltage or the excitation flux and by introducing additional resistance into the armature circuit.

Increase of the voltage during the starting process is achieved by switching the storage batteries from parallel to series connection, or with the aid of a start control box which increases the voltage smoothly.

When the storage batteries are connected in parallel, the starting system voltage is 24 - 27 V, while after switching to series connection, the voltage doubles. Such a voltage change is termed a step change. This change permits more rational use of the energy source capacity. The result of switching the storage batteries from parallel to series connection is an increase of the starter excess torque and an increase of the engine rotor speed, which makes the start more reliable.

Increase of the line voltage during the starting process with the aid of a start regulator connected to the excitation winding of the generator serving as the starter power source is multistep, i.e., smoother. Moreover, in this case, there is the possibility of applying a higher voltage to the starter terminals (up to 60 V), which also shortens the engine rotor acceleration time.

The engine rotor speed can also be increased by introducing additional resistance into the excitation winding circuit. An additional resistance is introduced into the armature circuit at the initial acceleration moment to provide shock-free engagement of the starter shaft with the engine rotor, and then this resistance is shunted and finally switched out of the circuit. This makes it possible to obtain a voltage of no more than 3 - 4 V across the starter terminals at the beginning of the acceleration, and then this voltage is increased.

Other techniques for increasing the voltage are also possible, for example, initial use of mixed excitation and then purely series or parallel excitation.

Piston Starters

Piston starters in the form of two-stroke air-cooled gasoline engines were used in the early stages of GTE development. They were quite complex in construction, had short service life, and high loading of all the engine components because of the small dimensions.

Compressor Turbostarters

These turbostarters are small high-speed GTE. They usually have centrifugal compressors, driven by a turbine. The excess starter turbine power is used to accelerate the rotor of the engine being started.

The advantages of these turbostarters include: high power (90 - 100 hp) with comparatively small starter size and light weight, repeated starts because of the small expenditure of electric energy and starting fuel. The disadvantages include long start duration (about two minutes), since the processes of starting the turbostarter itself and then the engine being started are performed sequentially. Moreover, the constructional complexity leads to lower reliability in comparison with the electric starters.

The compressor turbostarters operate on gasoline or kerosene. The use of gasoline leads to operational complications (a different fuel and separate fuel system are required). Engines similar to the turbostarters are also used in turbogenerator units to drive a dc generator which acts as the power source for the ship's system and electric starters.

Compressorless Turbostarters

Depending on the working medium which acts on the starter turbine, these turbostarters are subdivided into air, hot-gas and cartridge types.

The air-driven turbostarters may operate on cold or preheated air. In the latter case, the volumetric air flowrate through the turbine increases, which increases the turbine power and prevents turbine icing, which can occur as the air expands and its temperature decreases markedly. The turbostarters operating on preheated air are termed hot-air starters.

The air turbostarters utilize compressed air which is stored in on-board or ground bottles or is supplied by an on-board gas turbine unit, the compressor of an engine which has been started, or by a ground compressor station. The favorable features of these turbostarters are simplicity, reliability, high power, short starting time, and operation at low temperatures. However, the starters together with the compressed air sources are quite heavy. The primary drawback of these starters is the difficulty in supplying a sufficient quantity of compressed air.

The hot-air turbostarters operate on compressed air stored in on-board or ground bottles. A special fuel line is provided for preheating the air. Fuel and compressed air are fed into a combustion chamber. An ignition system provides the initial operation of the igniters. The gas temperature in the combustion chamber reaches 2000° C. The advantages of these starters are high power (300 - 500 hp), short starting time (to five seconds), and light weight; the disadvantages are the dependence on the compressed air sources and the requirement for providing cooling of the combustion chamber and insulation of the combustion chamber from the components of the engine being started.

The hot-gas turbostarters utilize steam, gas, or hot-gas obtained as a result of combustion, decomposition, or chemical reaction of various substances. The following substances can be used to generate the working medium which rotates the turbine:

mixture of combustion products of a hydrocarbon fuel and superheated water vapor;

steam-gas, obtained by decomposition of hydrogen peroxide;

gas obtained by chemical reaction between liquid fuels and an oxidizer or decomposition of a monopropellant.

The turbostarter operating on a mixture of the combustion products of a hydrocarbon fuel with superheated water vapor is a hot-air turbostarter with the addition of a system for injecting water into the combustion chamber. As the water vaporizes it reduces the gas temperature at the turbine inlet and increases the gas flowrate through the turbine and the turbine power. However, these starters require a reservoir for the water and a device for feeding the water by pump or compressed air. This increases the weight. Under low-temperature conditions the water must be warmed, and this complicates the operation.

In the turbostarters which use steam-gas obtained by the decomposition of highly concentrated (80 - 90%) hydrogen peroxide H_2O_2 with the aid of a catalyzer — potassium permanganate $NaMnO_4$, use is made of steam-gas generators in which high-pressure steam-gas (20 - 25 atm) is generated. The advantages of these starters are high power, light weight, comparatively low steam-gas temperature (400 - 700° C), and the possibility of repeated starts. However, the high freezing point of hydrogen peroxide (-10° C) and its high explosion hazard make the application of this starter difficult.

The turbostarters operating on gas obtained by chemical reaction between a liquid fuel (furfural alcohol) and an oxidizer (nitric acid) constitute a liquid rocket engine with utilization of the energy of the exhaust gases in the starter turbine (liquid gas-generators). These starters have high power and capability for repeated starts. The disadvantages include the operational complexity owing to the fire hazard in charging and storing the propellant components, and the necessity for a source of compressed air to feed the components.

For the turbostarters operating on gas obtained by decomposition of a monopropellant, containing both the fuel and oxidizer, we can use, for example, isopropyl nitrate $C_3H_2ONO_2$. This complex ester of isopropyl alcohol and nitric acid is nonpoisonous, nonexplosive, and insensitive to shock. In the combustion chamber the isopropyl nitrate mixes with air, forming a combustible mixture which then ignites. When the gas temperature reaches about 1000° C, the decomposition process begins to proceed by itself. The turbine inlet gas temperature does not exceed 450° C. This starter does not require high-temperature strength of the combustion chamber components and has a long service life. The starter power is high (up to 400 hp), and the starting time is short (4 - 15 seconds). A drawback is the complexity of the starting system. Sufficient capacity of the fuel tank is required for repeated starts.

The <u>cartridge starters</u> utilize the energy of powder enclosed in cartridges and ignited by a spark. The turbine inlet gas temperature reaches 1700 - 1900° C. The starter power is high (300 - 400 hp) with very light weight. The time required to accelerate the engine being started is very short (2 - 3 seconds). In spite of these advantages cartridge starters have not been widely used because of the high gas temperature, impossibility of manual interruption of the start, explosion hazard, and reduced power at low temperatures (combustion rate decreases).

Energy Sources for Electric Starters

Ground and on-board power units are used to supply energy for the electric starters. The on-board sources are storage batteries or dc generators, whose rotors are driven by the turbogenerator unit jet engine.

Aircraft storage batteries have a voltage of 24 - 27 V, electrical capacity 28 - 56 A·hr, and weigh about 30 kgf. A large number of batteries (no less than 20) is required to start a high-thrust (power) GTE. This leads to considerable weight (over 600 kgf) and difficulties associated with daily provisioning of the aircraft with charged batteries. It is also difficult to locate the large number of batteries aboard the aircraft. Therefore, there has recently been a trend to avoid the use of storage batteries as electrical energy sources for the starters installed on high-thrust (power) engines and to use a dc generator, whose rotor is driven by the turbogenerator unit jet engine.

The turbogenerator unit (APU) makes it possible to provide autonomous power supply for the electric starters, and the voltage developed during starting may be raised to 60 V. This provides reliable starting of high-thrust (power) TJE. The weight of the APU together with the fuel it consumes is considerably less than the weight of storage batteries of equivalent power.

A drawback of the APU is the overall increase of the time for starting the aircraft engines because of the time required to first

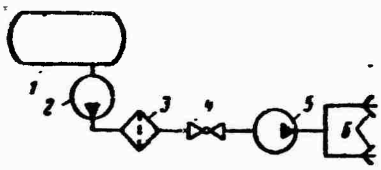


Figure 222. Starting fuel system: 1 - tank with starting fuel; 2 - boost pump; 3 - filter; 4 - cock; 5 - engine starting pump; 6 - starting nozzles.

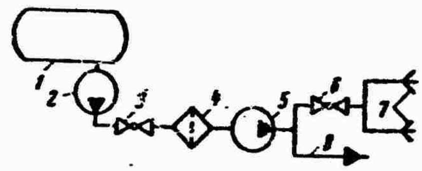


Figure 223. Starting fuel circuit: 1 - tank; 2 - airframe boost pump; 3 - fire shutoff valve; 4 - filter; 5 - engine boost pump; 6 - starting solenoid valve; 7 - starting nozzles; 8 - continuation of main fuel circuit line.

of GTE. A low-voltage ignition system with an electro-erosion surface-discharge igniter, which yields more thermal energy for igniting the fuel-air mixture, has recently come into use on GTE. The voltage applied to these igniters amounts to 1200 - 2500 V.

This igniter does not have a spark gap as in the normal high-voltage igniter. The central and side electrodes form an annular space which is filled with a ceramic insulator. The current is applied to the central electrode, and the side electrode makes electrical contact with the compound.

When current is applied to the igniter, an atomized layer of the electrode material is deposited on the end surface of the ceramic insulator. After this the end surface of the insulator becomes a semiconductor and during igniter operation serves as a source for the formation of a powerful and stable spark over the entire end surface of the ceramic insulator at a comparatively low voltage.

During operation of the combustion chamber the electro-erosive layer gradually burns up, and the igniter must be operated without combustion of fuel (igniter reconditioning) in order to restore the layer. This is achieved by applying current to the igniter during the start for some time prior to the moment fuel enters the starting nozzles (prereconditioning) or by cutting off the fuel supply to the starting nozzles first after termination of the start and then later cutting off the current to the igniters (post-reconditioning).

For flight safety the ignition circuits must be electrically independent of all the other electrical circuits. The high-voltage wires are run separately in shielded flex hoses and manifolds.

Operations

Prior to starting the engines, a check is made for proper operation of the powerplant systems and components and also the instruments used to monitor powerplant operation. The extent and details of these operations are defined by the corresponding maintenance regulation. The ramp, airplane, powerplants, power supplies, and cockpit must be prepared for the start.

Starting of the modern engines is automated, and the personnel starting the engines need only depress the "Start" button after performing all the preparatory operations. During the start the oil and fuel pressures, exhaust gas temperature, current, voltage, rpm, and time to reach the individual start stages and idle rpm are monitored.

In some cases provision is made for correcting the exhaust temperature of the engine being started by means of special devices (fuel "Trim" button). If necessary, the start can be aborted by means of shutoff devices.

If the engine accelerates to idle rpm in the established time without exceeding the allowable temperature, this is termed a normal start. An engine start in which the fuel ignites but the engine does not accelerate to idle rpm is termed an unsuccessful start.

A start made no sooner than two hours after engine shutdown is termed a cold start, and one made less than 15 minutes after engine shutdown is termed a warm start.

A simulated start with the ignition off but with fuel supplied to the engine is termed a false start, and a start without supplying any fuel is termed motoring. A false start is made when depreserving an engine, and motoring is performed after an unsuccessful start.

CHAPTER 12

ENGINE CONTROL

In the modern aircraft powerplant we must regulate simultaneously several parameters which are mutually related and depend on the flight speed and altitude, ambient temperature, power required, and other factors. The tendency toward maximal relief of the pilot's workload has led to the design of engine control accomplished by a single lever which controls the fuel flowrate. On most modern engines the trimming of the desired engine operating regime on the basis of the ambient conditions, limiting of the engine rotor speed and exhaust temperature, and also the starting program are automated.

Both manual and remote electrical control of aircraft powerplants are used. Manual control is used for the engine operating regime and reverse thrust control, while electrical remote control is used for controlling the systems (starting, fuel, oil, cooling, fire protection, and anti-icing). The questions associated with the control of these systems have been examined in the preceding chapters. In the present chapter we shall consider engine control.

Requirements

1. Engine control must be accomplished by a single lever, acting on the fuel flowrate. Optimal engine operating conditions must be provided in all engine regimes.

2. Each engine must have a separate control.

3. The engine control lever must occupy the same position for any given engine operating regime at different aircraft flight speeds and altitudes.

4. It is advisable that the control levers of several engines be positioned so that it is possible to control a single engine or all engines simultaneously. In controlling all the engines simultaneously, adequate synchronism of their operation must be ensured, since the presence of asymmetric thrust (power) leads to the appearance of a dangerous yawing moment, particularly during the takeoff run.

5. Smoothness of control. As the engine power control lever is displaced through its entire operating range there must not be any abrupt changes of engine thrust (power), which make piloting difficult.

6. In their extreme positions on the console, the engine control levers must not reach the end of the slot in the console covers. The differences of the clearances between the control levers and the ends of the slots in the covers in the extreme lever positions must not exceed the clearances given in the flight operations manual.

7. Provision must be made in the control system for components which prevent accidental or dangerous movements of the control organs. This protection is provided either by the use of protective covers installed over switches, buttons, cocks, or by means of special locks (detents, latches, and so on).

8. Control of the PE and TPE must provide for the possibility of reversing the propeller to brake the airplane during the landing roll.

9. The engine control levers must retain any set position. In the released condition they must displace with an effort of no more than 3 - 5 kgf.

10. The engine control cabling disconnects must be coordinated with the general aircraft disassembly scheme.

Engine Operating Regime Control Schemes

Engine control is a process in which each power lever position change corresponds to a quite definite throttle valve position change. Rotation of the throttle valve arm alters the amount of fuel entering the combustion chamber and therefore the engine operating regime.

Since the required fuel flowrate varies over a wide range with flight altitude and speed, control of the fuel flowrate with the throttle valve alone is difficult. For example, at altitudes above 9000 m the fuel flowrate at rated power is nearly equal to the flowrate at idle power on the ground. Another complication is that a fixed position of the control lever must be maintained with changes in flight altitude and speed in order to simplify operation of the engine. Trimming of the fuel flowrate with change of flight altitude and speed is accomplished by special devices which react to change of the ambient flight conditions.

Thus, in the case of manual engine control the pilot sets the fuel flowrate by changing the throttle valve flow area, while the programmed correction is performed by a regulator with a sensitive element which reacts to the compressor inlet air temperature and pressure. Thanks to its simplicity and operational reliability, this system is widely used on the modern TJE. Control of the TPE is also accomplished with the aid of a single lever which affects the fuel flowrate, and with the aid of an automatic rotational speed regulator which acts on the propeller pitch.

Shutdown of some engines is accomplished by a special lever (cutoff), linked with the fuel shutoff valve (cock). In those cases when engine control and engine cutoff are combined into a single lever, provision is made for protective devices which permit moving the power lever into the "Cutoff" position only after actuating these devices.

One of the characteristic features of helicopter powerplant control is the interconnection between the engine operating regime and the helicopter flight regime. It is well known that the helicopter lift force depends on main rotor angle of attack and the square of the blade speed relative to the airstream. With increase of the blade angle of attack there is an increase of the blade drag; therefore the engine power must be increased to maintain the given rotor speed. This requires definite coordination between the main rotor collective pitch setting and the engine control lever.

On most modern helicopters the throttle valve control is coordinated with the main rotor collective pitch control. In the cockpit there is a lever, called the collective-throttle, whose movement alters the position of the throttle valve and causes a corresponding change of the main rotor blade pitch. To provide a more precise setting of the engine operating regime, there is on the end of the manual collective-throttle control lever a rotating knob which can be used to regulate the degree of throttle opening regardless of the rotor pitch angle. Along with combined collective-throttle control, provision is made for separate control of the engines, which permits engine testing without changing the main rotor collective pitch.

In some helicopter designs, manual control of the collective throttle is replaced by a main rotor speed governor, similar to the PE crankshaft or TJE rotor speed governor. When the throttle is moved, this governor may be adjusted to a different rotational speed or may maintain the previously set speed. When automatic control is used, the possibility of manual control of the main rotor pitch is usually retained by providing for disconnect of the automatic control.

The engine control components include levers, cables, rods, bellcranks, brackets, pulleys, and so on. Depending on the aircraft type, engine control may be accomplished using push-pull rods or cables. The push-pull rods are made from 12 - 16-mm-diameter tubing and are equipped with clevis rodends, used for interconnecting the tubes with one another and also with the bellcranks and intermediate support brackets. Special end fittings are installed if the assembly conditions are such that tube length adjustment is required.

Engine control on multiengine aircraft is most frequently provided by use of flexible cables, consisting of two cables working in tension. The cables are multistrand with diameter ranging from 1.5 to 4 mm, with 2.5-mm-diameter cables being used most frequently. Pulleys of diameter equal to 20 or more cable diameters are installed at points where the cables bend. It is recommended that textolite pulleys with pressed-in ball bearings be used to reduce cable wear. The standard cabling terminals have identification markings. The cables which are in tension when engine power is increased are denoted Г1A, Г2A, Г3A, and so on, while the cables which are in tension when engine power is reduced are labeled Г1B, Г2B, Г3B, and so on. When connecting the cables by means of turnbuckles, their terminals must have the same identification.

Engine control may be accomplished from a central console (Figure 224) located between the pilots (Il-18 and An-24 airplanes), or from pilot's and copilot's consoles (Tu-104, Tu-124, and An-10 airplanes). The control levers located on the left and right consoles

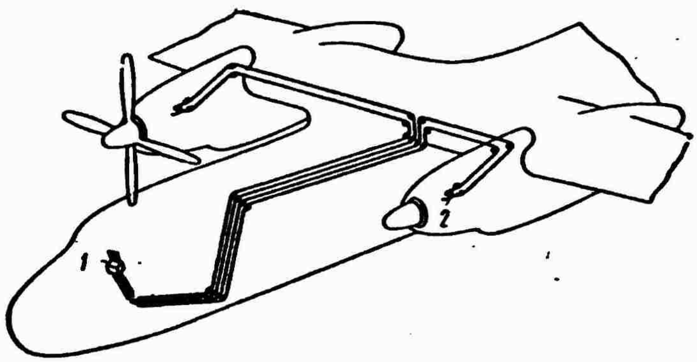


Figure 224. Engine control from central console:
1 - engine control levers; 2 - throttle valve lever.

are interconnected to provide the capability for engine control by either pilot. We see from these figures that the engine control levers are connected with the throttle valve levers arms by a system of cables, rods, and arms. The control cables run from the console levers to the pressurization bulkheads, and are then directed to the rig. t and left powerplants. The cabling is attached to end pulleys, i.e., pulleys connected with the control levers and pulleys installed near the powerplant. From this point on, the engine controls consist of rigid push-pull rods which transmit the motion through an arm on the end pulley and a push-pull rod to the throttle valve.

The engine control cables may be run along one or both sides of the fuselage. The direction of the cables and the spacing between them along the entire run are provided by pulleys and textolite fairleads. Turnbuckles are located in the most accessible areas to permit disconnecting the cabling or adjusting its tension. The engine control levers can be locked in any position by a special brake, whose lever is usually located alongside the engine control levers.

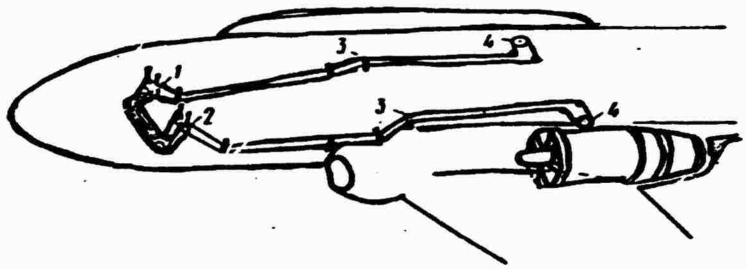


Figure 225. Engine control from left and right consoles:

1 - right console; 2 - left console; 3 - guide pulleys; 4 - end pulleys.

Each TPE control lever (Figure 226) includes a latch mechanism 2 and stops 7 installed on the console lever cover. When the power lever is retarded, the latch stops the movement at the "Flight Idle" power setting stop. The purpose of this stop is to indicate to the pilot that negative thrust will develop if the engine power is reduced further in flight. To move the power lever to the "Ground Idle" position, the pilot must withdraw the latch by applying a force to the finger 4. Similar stops are provided on the TJE power levers to prevent accidental movement of the power lever into the "Cutoff" position.

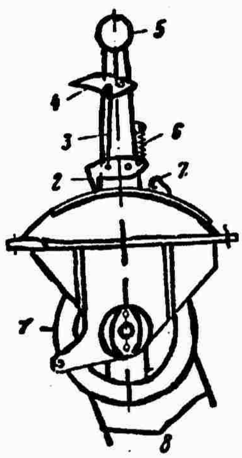


Figure 226. TPE control lever:

1 - pulley; 2 - latch; 3 - rod; 4 - latch finger; 5 - control lever handle; 6 - spring; 7 - stop; 8 - cables.

The engine control levers can be interconnected with the aircraft flight control system (Figure 227) so that when the flight controls are locked the engine power levers are limited to the

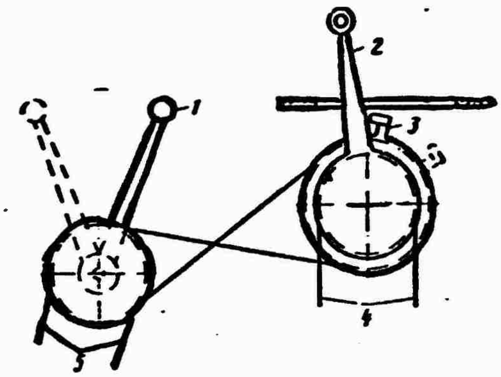


Figure 227. Interlock between control surface lock and engine controls:

1 - surface lock lever (controls unlocked in left position, locked in right); 2 - engine control lever; 3 - limiter; 4 - engine control cables; 5 - control surface lock system cables.

"Cutoff" or "Idle" position in order to prevent takeoff with the flight controls locked. Along with locking of the control surfaces, provision is also made for a special warning system which gives an audio signal when piloting errors are made (for example, when the power lever is advanced to takeoff power with a flap position not corresponding to the takeoff setting, or when the engine control power levers are moved to the "Idle" position in flight with the landing gear up).

Let us consider as an example a helicopter engine control scheme (Figure 228). The main rotor collective pitch and engine operating regime are changed by the collective-throttle levers 1, which are kinematically linked with the tilt control and the engine fuel supply system. The power levers are mounted on a special shaft. On this same shaft there is mounted a friction clutch with electro-hydraulic control. The friction clutch holds the collective-throttle lever in any position in order to provide continuously variable main rotor collective pitch settings. Normally the friction clutch is tightened by means of a knob so that the lever moves with a force of 20-25 kgf. The friction is disengaged by depressing a special button mounted in the upper end of the lever.

Individual control of the engines is provided by the levers 2. Both levers have friction clutches, which are set so that a force of 3 - 3.5 kgf is required to move either lever. The lever is retained against accidental movement by a tooth which engages with a detent in a rack. The lever is unlocked by depressing a button mounted on the lever. Upward movement of the lever increases

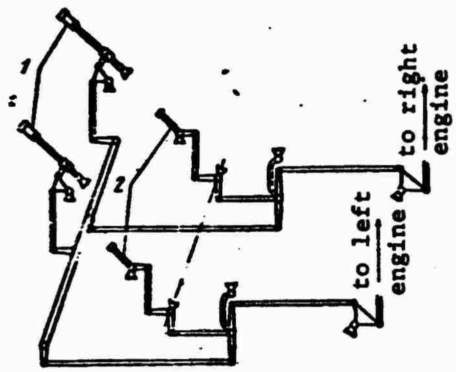


Figure 228. Helicopter engine control scheme.

engine power and downward movement decreases the power. The entire engine control system consists of push-pull rods and bellcranks. The rods, generally aluminum tubing, are necked down at the ends over a length equal to one diameter for the installation of fittings. The lengths of some of the rods can be regulated by threaded fittings.

Mechanical engine control has several drawbacks, the most typical being increase of the forces to overcome friction in the pressurized bulkhead fittings and change of the control system linear dimensions as the aircraft heats up at high flight speeds. All this leads to the need for developing remote (usually electric) engine control. Remote control is particularly advantageous when controlling several engines by means of a single lever, and also when controlling a single engine from several consoles.

Remote engine control can be obtained by using a servo electric drive (Figure 229). The lever 1 is rigidly linked with the transmitter shaft and the output channel of the actuator is rigidly linked with the fuel controller throttle valve. The connection between the other components is electrical. Induction potentiometers are used as the transmitters. The actuator is a unit in which there are mounted an electric motor, reduction gearing, and manual control limit switches. To increase operational reliability the control can be duplicated by means of a second servo system channel and manual control. Moreover, the control system may have a failure warning device which indicates system failure and the necessity to switch to the backup control system.

The development of supersonic airplanes requires automation of the engine control processes following a given flight program based,

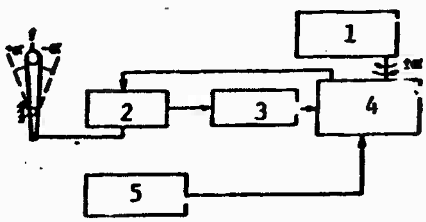


Figure 229. Remote engine control block diagram:

1 - fuel controller; 2 - transmitter; 3 - amplifier; 4 - actuator; 5 - manual control.

for example, on obtaining maximal flight range, maintaining the maximal cruising speed, and so on. For these purposes it is necessary to develop a separate automatic engine control system which is not related with the automatic engine governing systems (Figure 230). The system consists of a computer which generates the engine control signal. The primary signals come to this system from the flight speed and altitude sensors. A flight program controller must be provided in the cockpit. In addition the pilot can provide inputs to the engine operating regime through the power lever 1.

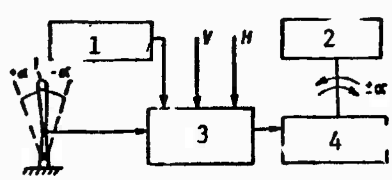


Figure 230. Automatic engine control block diagram.

1 - program controller; 2 - fuel controller; 3 - computer; 4 - actuator.

Thus the computer inputs include, in addition to the signals proportional to the flight parameters, the programmer signal and the command signal from the power lever. The use of such systems makes it possible to automate powerplant control, facilitate the aircraft piloting process, and ensure optimal flight regimes.

The thrust reversing control system is basically the same as the system for mechanical control of the engine operating regimes.

Control of Contingency Engine Operating Regime

On certain types of twin-engine aircraft provision is made for augmented operation of one engine in order to continue the takeoff if the other fails. This engine operating regime is termed max contingency operation. Control of the contingency regime provides for transient operation (a few minutes) of the engine at the maximal possible thrust (Figure 231).

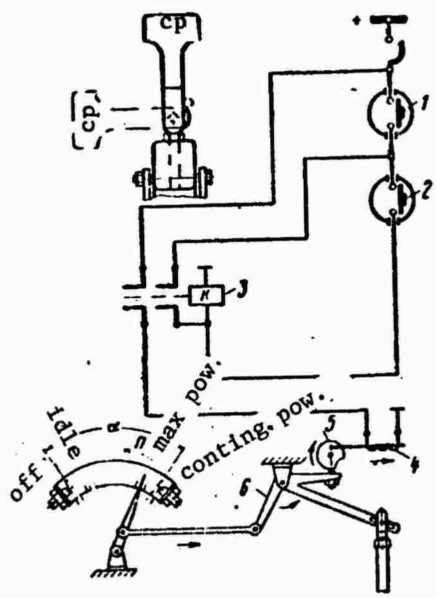


Figure 231. Schematic of control for contingency engine power operation.

The contingency power control mechanism is located on the pilot's control console ahead of the engine power levers. This mechanism consists of a housing on which a trigger bar is mounted. In the housing are mounted the switches 1 and 2. The trigger bar may be rotated 90° outboard (shown dashed). In this position of the trigger bar switches 1 and 2 are open. Prior to takeoff, the trigger bar is placed in the operating position, i.e., rotated 90° relative to the cruise position, which leads to closure of switch 1. In order to activate engine contingency control, the trigger bar must be deflected aft to close switch 2 and energize relay 3. When the contacts of relay 3 close, current is supplied to the winding of the contingency power limiter solenoid 4. The solenoid plunger rotates the limiter cam 5 into a position which permits the three-armed bellcrank 6 to deflect and increase the engine rotor speed above the maximal value as the power lever is moved full forward (into the contingency position), i.e., contingency power is activated. Current to the winding of relay 3 is terminated when the switch 1 is opened as the trigger bar is returned to the cruise position.

Use of contingency power is permitted only once for no more than two minutes, after which the engine must be removed from the airplane and sent to overhaul. The engine rotor speed (rpm), exhaust gas temperature, reason for using contingency power, and duration of its use are entered in the engine logbook. Contingency power is not used at ambient air temperatures below minus 15° C, since the rotor speed and engine thrust will be the same in the maximal takeoff and contingency regimes because of limited fuel system output.

Maintenance

Control system maintenance includes inspection of the cabling, pulleys, bellcranks, push-pull rods; and the hinges, cable segments at the pressure bulkheads, and the textolite fairleads are relubricated. Cable wear takes place primarily in the areas where the cables bend around pulleys and fairleads. Inspection includes checking for kinks, breaks in individual strands, and fraying of the cables. Another indication of cable wear is reduction of its diameter without breaking of strands and increase of brittleness because of work hardening. The latter is detected by breaking of the individual strands as the cable is flexed. Detection of any of these problems requires replacement of the cable.

Control rigging, security of the connections, and travel of the engine control levers are checked at the times specified in the maintenance manual. Rerigging is performed as necessary. Cable tension is regulated by means of turnbuckles. In so doing it is necessary to make certain that the threaded rodends are screwed in to the same length and that no more than three threads extend beyond the turnbuckle body.

Adjustment of the correspondence between the power lever position and the indicator on the fuel controller dial is accomplished by cable rigging and also by changing the length of the rods and levers in the rigid engine control components. The indication of the pointer on the fuel controller dial must correspond to the power lever position on the console (in addition, for some powerplants the fuel lever position indicator must also be aligned).

Cable tension changes with ambient temperature change because of the difference of the linear expansion coefficients of the cables and the aircraft structure. In this connection it is necessary to check the cable tension and readjust the tension if necessary. The tension of each cable is adjusted to a value determined by the ambient temperature, using special curves.

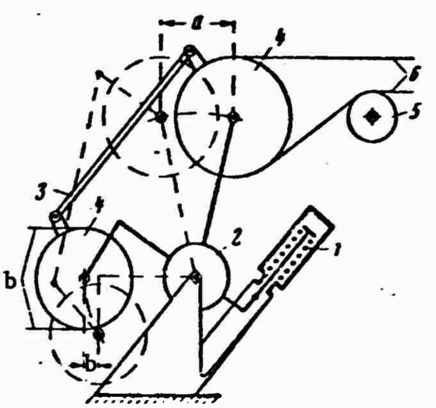


Figure 232. Cable compensator schematic:

1 - spring; 2 - pivoting mechanism; 3 - push-pull rod; 4 - end pulleys; 5 - guide pulleys; 6 - cables; a, b - pulley travels.

On some aircraft types special thermal compensators (Figure 232) are used in the engine control cabling to prevent cable sag as the airframe contracts at low ambient air temperatures. As the cabling becomes slack the system rotates (shown dashed) under the influence of the spring 1, thereby maintaining constant cable tension.

The engine control is rerigged when an engine is changed and also after partial or complete disassembly of the control system.

CHAPTER 13

AUTOMATED MONITORING SYSTEMS

Purpose. Requirements

The problems of ensuring flight safety and reliable operation of aircraft equipment in the course of the established service life of this equipment has become particularly important in connection with the equipping of civil airlines with new and more complex equipment. The methods presently used to monitor the technical status of aircraft and their powerplants are not entirely satisfactory, since they do not permit obtaining objective data on the condition of the component or system in question, and they require the expenditure of considerable time and use of highly qualified servicing personnel.

The continuously increasing complexity of aircraft equipment makes it impossible for the technician, without using special monitoring methods, to evaluate in a short time period, from the indications of the large number of instruments and warning lights located in the cockpit, the values of the parameters of the various systems, and on the basis of these data establish the status of the aircraft and the degree of its readiness for flight. It is particularly difficult to monitor the changes of the numerous parameters in the transient powerplant operating regimes.

The technological processes which exist at the present time for the maintenance of aircraft are basically the simple sum of the operations performed in maintaining the individual components and systems. The operational status of some systems is characterized by parameters whose measurement requires complex and expensive instruments. This situation had led to the appearance of a large amount of monitoring and metering equipment and specialized support equipment. The use of this equipment requires considerable time and highly qualified personnel.

The desire to maintain a given level of aviation equipment reliability as its design becomes more complicated leads to an increasing volume of maintenance operations. This leads to increase of the number of personnel involved in maintenance, and increase of the time and manhours expended in preparing the aircraft for flight.

New monitoring methods are required to provide high-quality maintenance of the aircraft and keep them in good condition. Use must be made of specialized equipment, including computers as an integral part of the system, to provide acceptable reliability of the monitoring results obtained. At the present time new systems, termed automated monitoring systems (AMS), are being developed and introduced into operation.

Aircraft powerplant maintenance is an integral part of the overall complex of operations performed in preparing an aircraft for flight, and therefore automation of powerplant operation monitoring must be perfected along with the development of the automated airframe monitoring systems.

The automated monitoring systems are designed to solve the following problems:

automatic status check of the basic aircraft and powerplant systems and output of quantitative check results;

shortening the time required to prepare aircraft for flight while providing high precision and reliability of the monitored parameters;

reduce the number of maintenance personnel and lower their required skill level;

automated trouble shooting;

reduce expenditure of aircraft equipment service life. In many cases automated monitoring avoids the necessity for running the engines to check the status of its individual components and systems;

provide prognosis of operating reliability of components, systems, and the aircraft as a whole.

These tasks to be solved by the AMS lead to the following requirements imposed on them:

the AMS must be developed along with the on-board equipment which they are to monitor;

use of the block principle in designing the circuits and constructing the AMS, i.e., the use of standard components which perform specific functions;

use of computer technology to analyze the information obtained during the check. This improves system performance, increases system reliability, and reduces equipment weight and size.

failure of AMS components, units, or blocks must not interrupt normal operation of the monitored systems;

universality of the AMS, i.e., possibility of using them for checking several types of flight vehicles;

high degree of automation, which eliminates monitoring subjectivity, shortens checking time, and reduces the number of maintenance personnel;

possibility of AMS or individual AMS element self-check;

reliability of the AMS must be greater than that of the system being monitored;

simplicity and convenience in operation. Possibility of AMS maintenance by specialists of average skill levels.

Classification

With regard to degree of universality the AMS are subdivided (Figure 233) into complex, complex-specialized, specialized, and

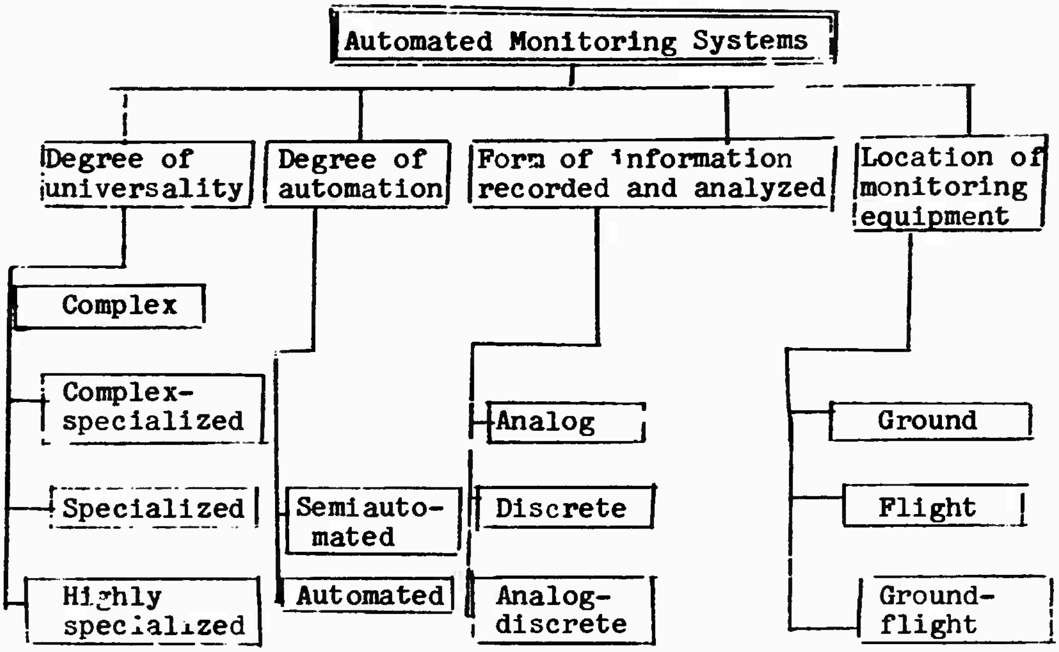


Figure 233. Classification of automated monitoring systems.

highly specialized [52]. The complex systems are used to monitor all types of aircraft and various forms of on-board equipment (of the various aircraft systems). The complex-specialized systems are designed to monitor the on-board equipment of aircraft of a single type. The specialized systems provide monitoring for individual systems of various aircraft types, and the highly specialized systems are used to monitor an individual system or its components for a single aircraft type.

With regard to degree of automation of the monitoring processes, the systems may be semiautomated and automated. The semiautomated systems include in addition to the conventional (standard) instruments, part of the equipment which performs automatically certain of the monitoring operations. The automated systems consist of devices which provide for automatic determination of the technical status of the unit being monitored. The role of the operator reduces to selecting the monitoring system operating regimes. If the checking process is accomplished without participation of the operator, such systems are termed automatic.

With regard to the form in which the information is recorded and analyzed, the AMS are subdivided into analog, discrete, and hybrid (analog-discrete). The first type is characterized by the fact that the information from the monitored unit is obtained, transmitted, and analyzed in continuous (analog) form. In the second type all the incoming information is handled in digital form. Specialized digital computers can be used for this purpose and make it possible to analyze and handle the incoming information in shorter time periods. Moreover, in the discrete systems information exchange between two or more subsystems can be accomplished with the aid of a single communications channel operating with a shift in time (cyclically), while in the analog systems a separate wire is required for the transmission of the signals from each parameter being recorded, which increases system weight and reduces reliability. The analog-discrete systems handle all the information in hybrid form.

We differentiate between ground-based, on-board, and ground-flight systems. In the ground-based systems only the sensors are located aboard the aircraft; the remaining equipment is located in special ground facilities. These systems, in turn, can be either stationary or mobile. In the mobile systems the equipment is installed in special trucks or other self-propelled vehicles, while in the fixed systems it is located in the airfield buildings. The connections between the stationary systems and the aircraft are accomplished using cable lines or radio links.

The on-board systems are those in which the analyzing and recording equipments are on board the flight vehicle. The on-board AMS provide minimal monitoring time, since they operate along with the unit being monitored and put out the required information immediately. In the ground-flight systems the basic test equipment is usually not a component part of the unit being monitored; therefore it can be used to monitor various types of components or several components of a given type.

The AMS can be used to conduct preflight, postflight, and periodic checks. In the systems designed for conducting preflight checks, the number of monitored parameters (monitoring depth) is not large, and the purpose of the check is to determine the readiness of the part (component, system) for use. The objective of the postflight checks is to determine the status of the part and to find any malfunctions which have developed in the part. More complex systems are required for this purpose, and the number of monitored parameters thus increases.

In addition to trouble shooting, the systems designed to conduct periodic checks are faced with the task of predicting the operation of the unit for a definite time period. These systems may provide qualitative monitoring with output of the monitor results in "go-no/go" or "high-normal-low" form or may provide quantitative monitoring. In the latter case the monitor results are presented in absolute, relative, or arbitrary units.

In addition to the characteristics examined above, the AMS can be classified on the basis of the form of the connection with the unit being monitored, the technique used to transmit and analyze the information, and the number of information channels. The selection of the type of system depends on such factors as the degree of complexity of the monitored unit, who is interested in the check results, the number and skills of the maintenance personnel, time expenditure on the check, and equipment costs.

Block Diagrams

Automatic monitoring systems of any type consist of two parts: the unit being monitored and the monitoring equipment (Figure 234). The monitored unit is any component, device, or system concerning the status of which it is necessary to obtain information during operation. The status of the unit is judged on the basis of the indications of the parameters being monitored. Since these parameters are of different natures, the system provides special sensors to transform the monitored quantity into an electrical signal which is convenient for amplification and remote transmission.

The monitoring equipment consists of the metering and indicating devices, which are designed for evaluating and analyzing the information taken from the monitored unit. Automation of the monitoring process provides for generation of test signals and their transmission to the monitored unit, measurement of the output signals, comparison of these signals with standard signals, analysis of the comparison results, and transmission of a signal to the programmer to continue or terminate the check.

Monitoring of the parameters which permit one to judge the status of a unit can be accomplished on units which are operating or not operating. In the first case test signals are not transmitted; rather the required information arises directly in the transducers during operation of the unit. If a nonoperating unit is being monitored, test signals must be generated at the input to the unit in order to obtain at the output signals characterizing the status of the unit which permit one to determine its operability. Depending on the type of parameters being monitored, these test signals may be of a different nature. They are usually transmitted in the form of voltage, current, pressure, angular and linear displacements, velocities, accelerations, and so on.

Special devices, called stimulating signal generators, are used to generate test signals of definite magnitude and shape. The primary purpose of the generators is to excite the part being checked on the basis of instructions obtained from the programmer. Reference signal generators are used in certain types of AMS. These generators are necessary to form signals corresponding to acceptable values of the monitored parameter. In the process of checking the part, these signals are compared with the normalized value of the parameter being monitored. Usually an attempt is made to combine the functions of the stimulating and reference signal generators into a single device.

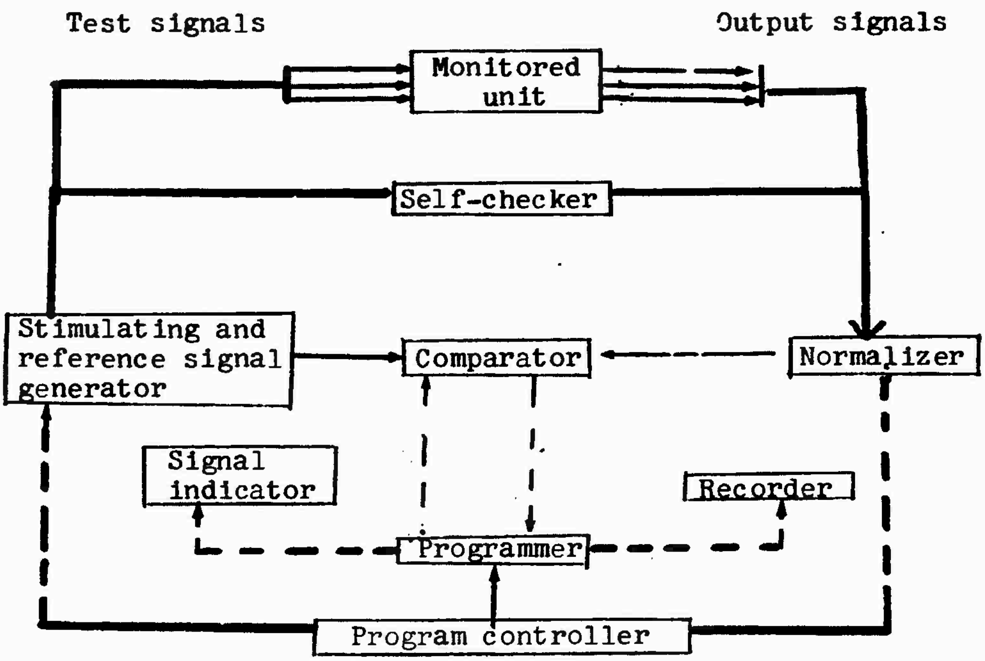


Figure 234. Block diagram of automated monitoring system.

Normalizers are used to reduce the values of the measured quantity to some prespecified value which can be measured easily by the other components of the system. The programmer determines the nature and sequence of the entire monitoring system operation. The programmer establishes the sequence of operations performed by the system during a complete check cycle, and for certain monitoring operations it carries out the required preparation of the circuits for output of the test signals and reception of the corresponding reactions. The programmer determines the limiting values of these signals and compares the monitored signal magnitude with its reference value.

The programmer consists of the program carrier, input and output commutators, devices for readout of the program instructions from the carrier, and also equipment for controlling the sequence of operations for forming the instructions by the stimulating and reference signal generator. This device operates in accordance with a program prepared in advance, which with the aid of special devices is entered on punched cards, punched tape, magnetic tape, or some other information carrier.

All the information from the monitored unit enters the metering-analyzing device (comparator). The latter obtains limiting-value signals from the programmer and the corresponding response signal from the system being checked, compares them with one another, and determines whether or not the response signal falls within its established tolerance. If the signal falls in the tolerance band, a "Go" rating appears on the signal display. If the signal does not fall in this range, a "No-Go" rating appears. In this case, provision is made for either switching the system to the next check step or stopping the system.

Recording equipment is provided in the AMS to provide information storage for subsequent statistical processing and analysis. The self-check unit generates a calibrated signal and transmits it to the comparator, where this signal is compared with the reference signal from the programmer. If the magnitude of the error obtained in comparing these signals does not exceed the acceptable value, the programmer puts out a command to transmit the check result to the signal indicator and recorder.

System self-check is usually performed prior to initiating monitoring of a unit or in the intervals between the primary check operations. In certain cases in which particularly high reliability is required, self-check may be performed prior to each monitoring operation.

Selection of Parameters Subject to Automated Monitoring

In resolving automated monitoring problems we encounter several difficulties, one of which is selection of the optimal number of parameters to be monitored. When analyzing the results obtained, a large number of measured parameters yields a clearer picture of the status of the unit being monitored. However, the number of such parameters cannot be increased excessively, since sensors must be installed to measure each parameter and this complicates the design of the unit being monitored, increases its weight, and reduces reliability. Moreover, the ground portion of the AMS becomes more complex.

The number of parameters should be selected so that the information obtained will be sufficient to characterize the operating status of the monitored unit.

There are several techniques for choosing the optimal number of monitored parameters. One method recommends taking as the primary parameters those which are normally checked during preflight preparation. Other methods propose the formulation of block diagrams with indication of all possible states of the monitored unit. On the basis of statistical data on the failure probability of the monitored unit, a calculation is made of the probability of the occurrence of each of these states. Then experimental or theoretical methods are used to establish the importance or effective weight of the monitored parameters for each of these states, i.e., we determine the list of parameters which are to be monitored.

Most acceptable is the technique whose idea amounts to the following. On the basis of analysis of the operation of the assemblies and systems, the plant being monitored is mentally broken down into block diagrams (for example, the powerplant is broken down into the fuel, oil, fire protection, and other systems). Then we construct cause-effect diagrams or logical sequence diagrams for trouble shooting on the basis of each of the selected monitoring parameters. In so doing we examine the points of the block diagram where the given parameter undergoes change.

Construction of the logical trouble shooting diagrams for a given block diagram makes it possible to determine the number of parameters to be included in the AMS and select the most rational points for locating the sensors. If a complex assembly (an engine, for example) cannot be broken down into block diagrams, it is divided up into structural units (for example, combustion chamber, turbine, and so on) with subsequent determination of the input and output parameters of each unit.

Analysis of these parameters makes it possible to identify which of them characterizes the operational status of the unit.

Using the oil system schematic shown in Figure 97 (see Chapter 4), we shall examine the technique for determining the number of parameters which should be included in the AMS. To construct the cause-effect relationship diagram (Figure 235), we make an analysis of the effect of various factors on system operation. It is well known that the magnitude of the oil pressure at the engine inlet depends on the operating condition of the (main) pressure pump and the pressure at the inlet to this pump. Since oil enters the pressure pump along two lines (from the radiator and from the tank through the makeup pump), the pressure at the pump inlet will depend on the condition of the makeup pump and its relief valve, and also on the condition of the assemblies installed in the scavenge manifold segment (radiator, air separator, pump scavenge elements).

The oil supplied to the engine for lubrication is filtered. If we monitor the pressure drop across the filter, we can judge the condition of the filtering elements and the degree to which they are clogged. Clogging of the filters in the oil scavenge line from the bearing cavities or failure of the scavenge pumps leads to overfilling

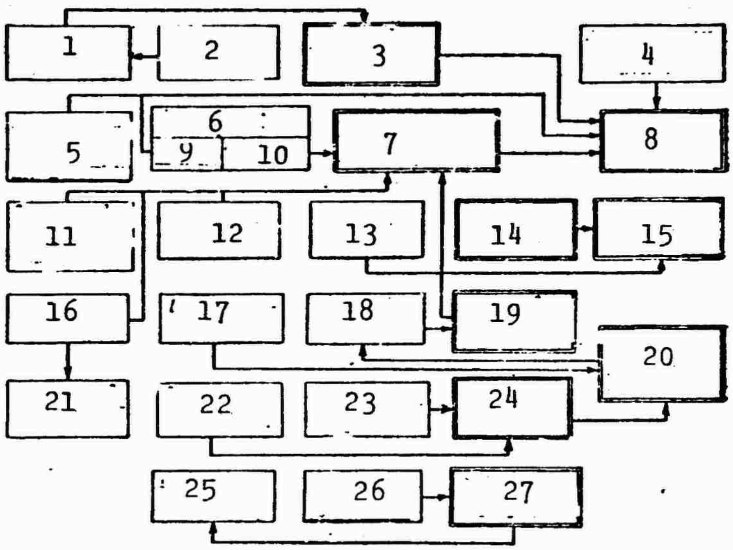


Figure 235. Cause-effect relationship schematic.

1 - filter condition; 2 - oil cleanliness; 3 - pressure drop across filter; 4 - condition of fuel controller oil system; 5 - condition of main oil pump relief valve; 6 - condition of main oil pump; 7 - main oil pump inlet pressure; 8 - filter outlet oil pressure; 9 - pressure section; 10 - scavenge section; 11 - condition of main oil pump relief valve; 12 - condition of makeup oil pump; 13 - condition of oil tank; 14 - pressure drop across labyrinths; 15 - oil level in tank; 16 - condition of air separator; 17 - condition of oil radiator; 18 - oil viscosity; 19 - pressure drop across radiator; 20 - main oil pump inlet temperature; 21 - oil tank pressurization pressure; 22 - condition of flap controller; 23 - condition of oil temperature regulator; 24 - radiator flap angle; 25 - oil level in bearing cavities; 26 - condition of rear bearing filters; 27 - scavenge pump oil-out pressure.

of these cavities with oil, which leads to leakage of the labyrinth seals and escape of oil into the engine gas and air flow passages. Increased oil consumption is also associated with failure of the labyrinth seals.

The degree of oil cooling in the PE and TPE powerplants depends on the position of the air-oil radiator duct flap. Therefore, we can take the flap angle as the monitored parameter and use the change of this angle to evaluate indirectly the proper operation of the oil temperature regulator.

Thus, in order to provide monitoring of oil system operation, in addition to the parameters presently used to evaluate the operating status of the system (engine inlet oil temperature and pressure, oil level in tank, and, on some aircraft types with PE and TPE the radiator duct flap angle as well), it is advisable to include in the AMS the following parameters: oil pressure at inlet to pressure pump, pressure drop across the filter, radiator, and labyrinth seals, and also the pressure downstream of the scavenge pumps.

Characteristic of powerplant system operation is the fact that their functioning is inseparably linked with engine operation. Therefore, an objective assessment of the reliability of these systems can be made only if their parameters are monitored on an operating engine. If we record the monitored parameters in flight, we can formulate a more objective description of the system operation, and we can also collect the data required for forecasting failure-free operation of the powerplant as a whole.

APPENDIX 1

CONVERSION TABLE

Quantity	Relation between measurement units
Force	1 kgf = 9.80665 N
Pressure	1 mm Hg. cm. = 133.322 N/m^2
	1 kgf/m^2 = 1 mm water. cm = 9.80665 N/m^2
	1 kgf/cm^2 = 1 at = $9.80665 \cdot 10^4$ N/m^2
Specific weight	1 kgf/m^3 = 9.80665 N/m^3
Work	1 kgf·m = 9.80665 J
Moment of a force	1 kgf·m = 9.80665 N·m
Power	1 hp = 735.5 W
Angular velocity	1 rpm/min = $\pi/30$ rad/sec
Kinematic viscosity	1 st = $1 \cdot 10^{-4}$ m^2/sec
	1 cst = $1 \cdot 10^{-6}$ m^2/sec
Temperature	t° C = T°K - 273.15
Quantity of heat	1 kcal = 4186.8 J
Specific heat	1 kcal/kg·deg = 4186.8 J/kg·deg
Heat flux	1 kcal/hr = 1.163 W
Thermal conductivity coefficient	1 kcal/m·hr·deg = 1.163 W/m·deg
Heat transfer coefficient	1 $kcal/m^2 \cdot hr \cdot deg$ = 1.163 $W/m^2 \cdot deg$

APPENDIX 2

STANDARD ATMOSPHERE (after GOST 4401-64)

Height *H*, m	Barometric pressure P			Temperature	
	N/m^2	mm Hg. cm.	kgf/m^2	*T*, °K	*t*, °C
0	101 325	760,00	10 332,3	288,15	15,00
1 000	89 876	674,12	9 164,8	281,65	8,50
2 000	7[illegible] 498	596,28	8 106,5	275,14	1,99
3 000	[illegible]25	525,98	7 150,8	268,64	— 4,51
4 000	61 656	462,46	6 287,2	262,13	—11,02
5 000	54 045	405,37	5 511,1	255,63	—17,52
6 000	47 213	354,13	4 814,4	249,13	—24,02
7 000	41 098	308,26	4 190,8	242,63	—30,52
8 000	35 648	267,38	3 635,1	236,14	—37,01
9 000	30 791	230,95	3 139,2	229,64	—43,51
10 000	26 491	198,70	2 701,3	223,15	—50,00
11 000	22 690	170,19	2 313,7	216,66	—56,49
12 000	19 391	145,44	1 977,3	216,66	—56,49
13 000	16 572	124,30	1 689,9	216,66	—56,49
14 000	14 164	106,24	1 444,3	216,66	—56,49
15 000	12 107	90,81	1 234,6	216,66	—56,49
16 000	10 348	77,62	1 055,2	216,66	—56,49
17 000	8 846	66,35	902,0	216,66	—56,49
18 000	7 562	56,72	771,1	216,66	—56,49
19 000	6 465	48,49	659,2	216,66	—56,49
20 000	5 527	41,46	563,6	216,66	—56,49
21 000	4 725	35,44	481,9	216,66	—56,49
22 000	4 040	30,31	412,0	216,66	—56,49
23 000	3 455	25,91	352,3	216,66	—56,49
24 000	2 954	22,16	301,2	216,66	—56,49
25 000	2 526	18,95	257,6	216,66	—56,49
26 000	2 162	16,22	220,5	219,40	—53,75
27 000	1 855	13,91	189,1	222,14	—51,01
28 000	1 594	11,96	162,6	224,87	—48,28
29 000	1 373	10,30	140,0	227,61	—45,54
30 000	1 184	8,88	120,7	230,35	—42,80

(continued on next page)

APPENDIX 2 (continued)

STANDARD ATMOSPHERE (after GOST 4401-64)

Density					Speed of sound a, m/sec	Kinematic viscosity $\nu \cdot 10^9$ m^2/sec
Mass ρ		Weight γ		Relative Δ		
kg/m^3	$kgf \cdot sec^2/m^4$	N/m^3	kgf/m^3			
1,22500	0,12492	12,0131	1,2250	1,00000	340,28	14 607
1,11170	0,11336	10,8950	1,11170	0,90751	336,43	15 812
1,00660	0,10265	9,8690	1,00660	0,82171	332,52	17 146
0,90941	0,92734	8,9172	0,90941	0,74237	328,56	18 624
0,81942	0,83558	8,0346	0,81942	0,66891	324,56	20 271
0,73654	0,75106	7,2196	0,73654	0,60125	320,51	22 103
0,66022	0,67324	6,4675	0,66022	0,53695	316,41	24 153
0,59010	0,60174	5,7810	0,59010	0,48171	312,25	26 452
0,52591	0,53628	5,1490	0,52591	0,42931	308,05	29 030
0,46712	0,47633	4,5699	0,46712	0,38132	303,78	31 942
0,41357	0,42172	4,0443	0,41357	0,33761	299,45	35 232
0,36485	0,37204	3,5657	0,36485	0,29784	295,07	38 966
0,31180	0,31795	3,0459	0,31180	0,25453	295,07	45 595
0,26648	0,27173	2,6007	0,26648	0,21753	295,07	53 351
0,22776	0,23225	2,2212	0,22776	0,18593	295,07	62 420
0,19467	0,19851	1,8976	0,19467	0,15891	295,07	73 029
0,16640	0,16968	1,6210	0,16640	0,13584	295,07	85 437
0,14224	0,14504	1,3846	0,14224	0,11611	295,07	99 952
0,12159	0,12399	1,1817	0,12159	0,09926	295,07	116 920
0,10395	0,10600	1,0091	0,10395	0,08486	295,07	136 760
0,08887	0,09062	0,8630	0,08887	0,07255	295,07	159 970
0,07598	0,07748	0,7404	0,07598	0,06203	295,07	187 100
0,06497	0,06625	0,6247	0,06497	0,05303	295,07	218 830
0,05555	0,05665	0,5381	0,05555	0,04535	295,07	255 930
0,04750	0,04844	0,4619	0,04750	0,03878	295,07	299 290
0,04062	0,04142	0,3942	0,04062	0,03316	295,07	349 980
0,03434	0,03501	0,3266	0,03437	0,02803	296,93	418 420
0,02909	0,02966	0,2785	0,02909	0,02374	298,78	499 110
0,02470	0,02519	0,2403	0,02470	0,02016	300,61	593 700
0,02101	0,02142	0,2020	0,02101	0,01715	302,43	705 100
0,01790	0,01825	0,1736	0,01790	0,01461	304,25	835 650

APPENDIX 3

SPECIFIC WEIGHT OF AVIATION FUELS, kgf/m^3

t, °C	Kerosene T-1	Kerosene TC-1	Kerosene T-2	Kerosene T-5	Gasoline B-70
−60	880	835	827	904	825
−40	865	821	812	890	807
−20	849	807	797	876	789
0	835	793	782	862	771
20	820	779	767	848	753
40	809	765	752	834	735
45	805	761	748	830	730
60	794	750	737	820	717
80	782	735	723	807	699
100	767	720	709	793	681
120	751	705	695	780	660
140	739	690	681	765	639

APPENDIX 4

SATURATED VAPOR PRESSURE OF AVIATION FUELS FOR VAPOR/LIQUID PHASE RATIO 4/1, mm Hg

t, °C	Kerosene T-1	Kerosene TC-1	Kerosene T-2	Kerosene T-5	Gasoline B-70
0	25	31	50	19	60
20	32	40	65	21	120
40	42	57	95	26	240
45	47	64	130	28	285
60	60	86	235	32	420
80	89	140	395	44	750
100	159	231	580	65	1300
120	220	382	820	98	—
140	429	618	1270	158	—

APPENDIX 5

KINEMATIC VISCOSITY COEFFICIENT OF AVIATION FUELS, cSt

t, °C	Kerosene T-1	Kerosene TC-1	Kerosene T-2	Kerosene T-5	Gasoline B-70
−60	31,5	11,8	9,6	—	3,2
−50	15,0	7,9	4,8	98,0	2,0
−40	8,6	5,0	4,4	43,5	1,7
−20	4,1	2,8	2,1	14,5	1,3
0	2,5	1,8	1,5	6,8	0,9
20	1,6	1,3	1,1	3,8	0,7
40	1,2	1,0	0,9	2,5	0,6
45	1,1	0,9	0,8	2,2	0,5
60	0,9	0,8	0,7	1,7	0,5
80	0,7	0,7	—	1,3	0,4
100	0,6	0,6	—	1,0	0,3
120	0,5	0,5	—	0,7	0,3
140	0,4	0,4	—	0,6	0,3

APPENDIX 6

SPECIFIC HEATS OF AVIATION FUELS, kcal/kg·deg

t, °C	Kerosene		Gasoline B-70
	T-1	T-5	
20	0,478	0,456	0,492
40	0,499	0,486	0,514
45	0,505	0,490	0,519
60	0,522	0,509	0,536
80	0,544	0,529	0,562
100	0,568	0,551	0,587
120	0,592	0,574	0,613
140	0,616	0,596	0,641

APPENDIX 7

THERMAL CONDUCTIVITY OF AVIATION FUELS, kcal/m·hr·deg

t, °C	Kerosene		Gasoline B-70
	T-1	T-5	
—50	0,109	0,106	0,113
0	0,103	0,102	0,103
50	0,096	0,097	0,095
100	0,090	0,092	0,068
150	0,083	0,088	0,079

APPENDIX 8

SIZES AND WEIGHT-PER-METER OF ALUMINUM AND ALUMINUM ALLOY TUBING (STEEL TUBING WEIGHS 2.8 TIMES MORE), kgf

Outside diameter, mm	Wall thickness, mm			
	1	1,5	2	2.5
10	0,079	0,112	0,141	0,165
11	0,088	0,125	0,158	0,187
12	0,097	0,139	0,176	0,209
14	0,114	0,165	0,211	0,253
16	0,132	0,191	0,246	0,297
18	0,150	0,218	0,281	0,341
20	0,167	0,244	0,317	0,385
22	0,185	0,270	0,752	0,429
24	0,202	0,297	0,387	0,473
25	0,211	0,310	0,405	0,495
26	0,220	0,323	0,422	0,517
28	0,238	0,350	0,457	0,561
30	0,255	0,376	0,4[illegible]3	0,605
32	0,273	0,402	0,528	0,649
34	0,290	0,429	0,563	0,693
36	0,308	0,455	0,598	0,737
48	0,325	0,482	0,633	0,780
40	0,343	0,508	0,669	0,825

(continued on next page)

APPENDIX 8 (continued)

SIZES AND WEIGHT-PER-METER OF ALUMINUM AND ALUMINUM ALLOY TUBING (STEEL TUBING WEIGHS 2.8 TIMES MORE), kgf

Outside diameter, mm	Wall thickness, mm			
	1	1,5	2	2,5
42	0,361	0,534	0,704	0,869
45	0,387	0,574	0,756	0,935
48	0,413	0,614	0,809	1,000
50	0,431	0,640	0,844	1,015
52	0,449	0,666	0,880	1,089
55	0,475	0,706	0,932	1,155
58	0,501	0,746	0,985	1,221
60	0,519	0,772	1,020	1,265
65	—	0,838	1,108	1,374
70	—	0,904	1,196	1,484
75	—	0,970	1,284	1,594
80	—	—	1,372	1,704
85	—	—	1,460	1,814
90	—	—	1,548	1,924
95	—	—	1,636	2,034
100	—	—	—	2,144
105	—	—	—	2,254
110	—	—	—	2,364

APPENDIX 9

PARTIAL PRESSURE AND HEAT OF VAPORIZATION OF SATURATED WATER VAPOR

t, °C	Partial pressure				Heat of vaporization, MJ/kg
	N/m^2		mm Hg, cm.		
	above ice	above water	above ice	above water	
—40	28,7	—	0,097	—	2,60
—30	38,2	—	0,286	—	2,58
—20	103,5	—	0,776	—	2,55
—10	260,0	286	1,950	2,147	2,53
0	610,0	610	4,579	4,579	2,50
10	—	1 228	—	9,209	2,48
20	—	2 340	—	17,940	2,45
30	—	4 210	—	31,800	2,43
40	—	7 370	—	55,300	2,40

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SYMBOL LIST

Russian	Typed	Meaning
В	p	propeller
С	j	jet
В	a	air
КР	cr	cruise
Р	roll	takeoff roll
ВЗЛ	to	takeoff
СР	av	average
ОТР	uns	unstick
ПОТР	req	required
Р	gr	ground run
расп	avail	available
С.У	pp	powerplant
ДВ	eng	engine
ГИР	gyro	gyroscopic
Э	op	operational
Д	e	engine
расч	des	design
аэр	aero	aerodynamic
кап	cowl	cowling
ПР	ult	ultimate
Ц.Т.	cg	center of gravity
СИМ	sym	symmetric
ОбР.СИМ	antisym	antisymmetric
П	s	suspension
К	m	mount
а	d	damping
СДВ	sh	shear
СМ	com	compression
Г	h	hydraulic
ТР	fr	friction
М	l	local
Г.Ш	flex	flex hose

Russian	Typed	Meaning
э	eq	equivalent
посл	ser	series
пар	par	parallel
маг	circ	circuit
и	i	inertial
вх	in	inlet
кав	cav	cavitation
н	p	pump
потр	req	required
пол	use	useful
т	f	fuel
н.о	un	unusable
исп	ev	evaporation
б	t	tank
к	eq	equipment
с	ex	free expansion
т	t	turbine
п	p	pump
пнд	EBP	engine boost pump
маг	line	line
пер	by	bypass
а.з.	af	accum. fill
пн	BP	boost pump
к	v	valve
а	acc	accumulator
з	fill	filling
от	op	opening
а.р	acc. dis	accum. discharge
п	use	useful
р	dis	discharge
опт	opt	optimal
изб	ex	excess

Russian	Typed	Meaning
Б	T	tank
ИЗБ	diff	differential
ПНЛ	ABP	airframe boost pump
ДОП	per	permissible
П.С	bl	boundary layer
Г.В	ha	hydraulic air
И.В	ia	inertial air
ЗАД	ass	assumed
К.Б	tv	tank-valve
К.А	av	accum. valve
В.А	aa	accum. air
Г.К	hv	hydraulic-valve
изб.доп	diff per	permissible differential
И.Т	if	inertial fuel
СТР	bl	bleed
над	press	pressurization
П	p	pickup
МАГ	man	manifold
М.Б	ot	oil in tank
М.С	os	oil system
расх	con	consumed
М	o	oil
ред	red	reduction gearing
ПОДШ	bear	bearings
НОМ	rat	rated
ОП	des	descent
пред	lim	limit
ГОР	hot	hot body
ТОПЛ	f	fuel
охл.тела	cold	cold.body
ТОРМ	st	stagnation
СМ,СТ	w	wall
ИЗ	ins	insulation

Russian	Typed	Meaning
ВыХ	out	outlet
Р	r	radiator
ОБД	cool	cooling flow
ОМ	disk	disk
ТР	t	tube
Ж	ar	area
ПОС	AIS	Anti-icing System
КОНВ	conv	convective
Н.В	hw	heating water
ИСП	vap	vaporization
ПОВ	sur	surface
В	w	water
В.ПР	a red	air, reduced
КР	cr	critical
ВХ	in	in
ВыХ	out	out
нагр	heat	heating
О.В	ext	extinguishant
ВОЗД	air	air
МАГ	line	main line
ОТВ	br	branch line
Н.Г	ig	inert gas
Ж	or	orifice
К	c	compressor
ВН	ex	external
Эф	eff	effective
П	t	total
ДОП	add	additional
В	e	exhaust
С	n	nozzle
В	e	exit
Р.С	ne	expan., nozzle
Р	r	reversal
СТР	j	jet

Russian	Typed	Meaning
НАИВ	opt	optimal
В	p	propeller
П	cw	counterweight
Т	fr	friction
ПОС	lnd	landing
ПР.УЛ	is	intermediate stop
В	th	throttle valve
полата	flt	flight
МИН	min	minimum
КОНТ РеГ	gov. cont.	gov. control
ФЛ	feath	feather
НОМ	nom	nominal
М.Г	idle	idle
З	s	set
СТ	st	starter
У	a	accelerating
Р	r	rotor
К	co	compressor
зап	start	start
Я	a	armature
Д	add	additional